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FINAL REPORT
ACQUISITION/EXPULSION SYSTEM FOR EARTH ORBITAL PROPULSION SYSTEM STUDY

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Lyndon B. Johnson Space Center
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FOREWORD

This document is submitted to the National Aeronautics and Space Administration, Johnson Space Center by Martin Marietta Corporation, Denver Division, as a part of the final report for Contract NAS9-12182, entitled *Acquisition/Expulsion System for Earth Orbital Propulsion System Study* and consists of the five following volumes.

Volume I - Summary Report;

Volume II - Cryogenic Design

Volume III - Cryogenic Test;

Volume IV - Flight Test Article;

Volume V - Earth Storable Design.

This work was administered under the technical direction of Mr. Larry R. Rhodes, Power and Propulsion Division, NASA-JSC, Houston, Texas. Mr. Howard L. Paynter, Chief of Thermodynamics and Fluid Mechanics Section, Propulsion Department, was the Martin Marietta Program Manager.

The following Martin Marietta personnel made significant contribution to the Phase C, Flight Test Article study effort.

Mr. Howard L. Paynter

Mr. Ralph N. Eberhardt

Mr. Richard P. Warren

Mr. Thomas R. Barksdale

Mr. Earl R. Wilson

Mr. Duane J. Brown

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SUMMARY

Two orbital test plans were prepared in this phase of the contract. The test plans were to verify one of the passive cryogenic storage tank/feedline candidate designs proposed in Phase A of the study.

One plan considered the orbital test article to be launched as a dedicated payload using an Atlas F/Burner II launching configuration. The second plan proposed to launch the orbital test article as a secondary payload on the Titan IIIE/Centaur Proof Flight scheduled for January 1974. The Titan IIIE/Centaur proof flight launching the test article as a secondary payload was estimated to cost $1.6M, approximately one-fifth the cost of using an Atlas F/Burner II. Therefore, the secondary payload concept was pursued until January 1973, when work to build the hardware for this phase of the contract was terminated for lack of a sponsor for the flight. Chapter III of this volume describes this secondary payload program plan in detail. Its counterpart, the dedicated payload launched on an Atlas F is described in Chapter II.

The passive DSL tank/feedline design has great potential application for space missions in the near future. The design should be validated in a test flight, which will provide an extended period (7 to 14 days) of low-g, at an early date.

I. INTRODUCTION AND BACKGROUND

The objective of the three phased program was to design and verify passive acquisition/retention devices for liquid propulsion systems for earth orbiting vehicles. Phase A of the program was limited to cryogenic propellants and Phase B to earth storables. The objective for this phase of the program, Phase C, was to develop an orbital test plan to verify the passive tank/feedline design selected in Phase A. Two program plans were formulated. One was based upon a dedicated launch using an Atlas-F either with, or without, an upper-stage. The second program plan was different in that the cryogenic test module was a secondary payload. The specific opportunity pursued under the secondary payload approach was the Titan IIIE/Centaur to be flight tested in January, 1974.

The schedule for the 23-month technical effort begun in August, 1971, is presented in Figure I-1. This phase of the study, "Flight Test Article," comprised three separate tasks (V, VI, and VII). The first task begun in April, 1972, Task V, was a four-month effort to develop the dedicated launch vehicle approach. This was the preferred approach at program initiation, as proposed by Martin Marietta in Ref I-1. The dedicated launch plan to place the cryogenic test module in earth orbit was submitted under MCR-72-196 dated July 31, 1972. It was reviewed and approved by NASA-JSC. The dedicated Atlas-F orbital program is presented in this volume under Chapter II.

Task VI was begun in May to design the dual-screen-liner (DSL) tank/feedline flight test article, based on the design recommended by Mr. G. Robert Page, Technical Director for Phase A, for NASA approval at the third quarter review held at NASA-JSC on May 26, 1972. The design was approved by Mr. Larry R. Rhodes, JSC Technical Monitor (Ref I-2).

In mid-August, the Titan IIIE/Centaur proof flight opportunity was recognized as a more attractive scheme than the dedicated launch for the following economic reasons: (1) the need for an upper-stage with Atlas-F was verified (Ref I-3), making the projected cost \$7-8M for the dedicated approach; (2) the decision that the Shuttle Orbiter would not use cryogenic propellants, at least for the initial versions; (3) in addition to the Orbiter, circa 1980-1985 vehicles, like the Space Tug, may not use cryogenic propellants; (4) the DSL test article being designed under Task VI could replace one of the dummy masses being flown as part of the payload

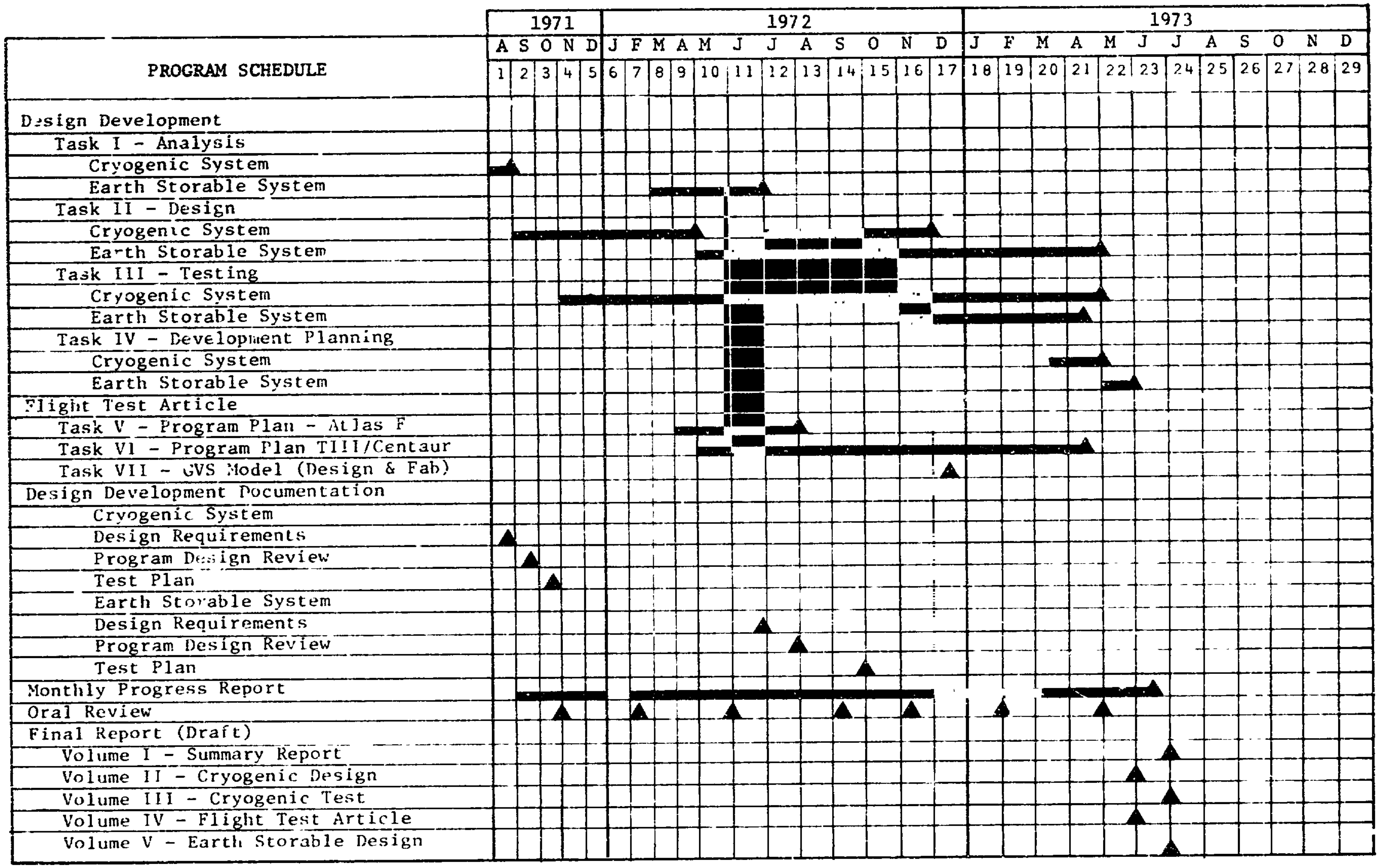

Figure I-1 Program Schedule

for the proof flight without impairing objectives of the proof flight; and (5) the cost to conduct the cryogenic orbital test in this manner would be low, approximately $1.6M. A meeting held with Ceneral Dynamics Corp., Convair Aerospace Division (GDCA) personnel (Ref I-4), showed that the secondary payload opportunity was probable. On September 6, 1972, Martin Marietta agreed to pursue this specific flight opportunity under Task VI with JSC approval (Ref I-5).

The proof flight schedule required ground vibration tests to be conducted on the payload at GDCA beginning January, 1973. Since our plan called for the orbital test module to replace part of this payload, it was necessary to deliver our module to GDCA for incorporation into the payload for the tests. Therefore a model of the orbital module had to be analyzed, designed, fabricated, and delivered to GDCA in January, 1973. Task VII was begun in late August to satisfy this requirement. Work under Task VII is summarized in Chapter IV.

The secondary payload orbital test plan, Task VI, is outlined in Chapter III. Sponsorship for this orbital test plan was not obtained (Ref I-6) and as a result pursuit of the proof flight opportunity was terminated in January, 1973. Task VI was continued to mid-April to finalize the program plan since it is representative of the piggy-back or secondary payload approach.

Conclusions and recommendations for the flight test phase of the program are presented in Chapter V.

II. PROGRAM PLAN - EXPERIMENT AS A DEDICATED PAYLOAD

This program plan was prepared for the cryogenic orbital test article program and is based upon a dedicated payload concept. Two different sized tank/feedline test articles were considered in the program plan. One considered the dedicated payload being launched by an Atlas-F with no upper stage (Ref II-1). This configuration was capable of placing a 76 cm (30 in.) dia spherical LO_2 test module into a 185 km (100 n mi) circular Earth orbit.

Concern over the hazard of the Atlas-F debris penetrating the atmosphere in an unpredictable manner upon orbit decay, shifted the analysis to a second Atlas-F configuration. This alternative launch vehicle was the Atlas-F with a Burner II upper stage. The 2-1/2 stage configuration would have a predictable impact downrange location for the spent Atlas-F because it would only reach a sub-orbital velocity at burnout. The Burner II and the flight test module would continue into a circular 185 km (100 n mi) Earth orbit.

The Atlas-F/BII configuration showed a payload capability of 1,580 kg (3500 lb_m) to this orbital destination (Ref II-2). Therefore, the dedicated payload orbital test module could now use this full capability, or place a smaller payload into a higher orbit altitude, or accomodate a secondary payload of some other mission. If full use was made of the 1,580 kg (3500 lb_m) capability for the cryogenic orbital test module, the LO_2 storage tank could be increased to 106 cm (42 in.) dia. This spherical tank would contain 678 kg (1500 lb_m) of LO_2.

These two Atlas-F configurations dominated the fleet of U.S. Launch vehicles because of economic considerations. The basic Atlas-F is available for its refurbishment cost ($0.9M) through the AFSC/SAMSO. The following section describes the launch vehicle selection procedure used in this program plan.

A. LAUNCH VEHICLE SELECTION

The most cost-effective launch vehicle selection was made after a survey of four families of launch vehicles. Configurations of the Scout, Delta, Atlas, and Titan families were evaluated. Table II-1 shows comparative performance and characteristic data for selective members of the families of launch vehicles.

Table II-1 Candidate Launch Vehicle Systems

LAUNCH VEHICLE	LAUNCH SITE	PAYLOAD CAPABILITY 185-km (100 n mi) ALTITUDE		AVAILABLE FAIRING DIMENSIONS			
				INSIDE DIA		INSIDE LENGTH	
		kg	lbm	cm	in.	cm	in.
Scout	Wallops Is.	235	520	96	38	254	100
Scout	WTR	201	445	96	38	254	100
Delta 303	ETR	1500	3300	145	57	254	100
Delta 904	ETR	1940	4300	145	57	254	100
Atlas F	WTR	590	1300	190	75	458	180
Atlas F/Burner II	WTR	1580	3500	190	75	458	180
Titan II	WTR	>1720	>3800	282	111	356	140+
Titan III BS	WTR	2850	6300	282	111	356	140+
Titan IIIE/Centaur*	ETR	3550	7850	396	156	1700	670

*Elliptical 23,200x34,300-km (12,515x18,515-n-mi) orbit.

The work led to a tentative selection of an Atlas-F as the launch vehicle. This vehicle has the capability to place 590 kg (1300 lb_m) into a 185 km (100 n mi) circular Earth orbit and can adequately accomodate the orbital test module requirement of 500 kg (1100 lb_m) to same destination.

The tentative selection of the Atlas-F, whose performance was adequate, was set aside about August 1, 1972 because of concern over the unpredictable impact location of Atlas-F debris (Ref II-3). The Atlas-F, with a Burner II upper stage, was then considered since the Atlas-F spent stage would impact in the Pacific Ocean.

The cryogenic orbital test module size can be increased to include a 106 cm (42 in.) dia spherical LO_2 storage tank containing 678 kg (1500 lb_m) of LO_2, as mentioned earlier.

The addition of a Burner II upper stage was estimated to cost approximately $2M (Ref II-4). Thus, the total cost for placing a 106 cm (42 in.) dia storage tank test module into orbit was approaching $7-8M. The latter is less than other competitive vehicles of the same capability. However, the $7-8M figure is about 5 times greater than launching the cryogenic orbital test module as a secondary payload on the Titan IIIE/Centaur proof flight. Cost estimates prepared by MMC to place the 76 cm (30 in.) dia storage tank test module on this launch vehicle showed a cost of $1.6M. The secondary payload launch scheme is presented and described in Chapter III.

The Scout launch vehicle is unable to orbit the 500 kg (1100 lb_m) orbital test module to the required 185 km (100 n mi) altitude and, therefore, was not considered further. The two Delta launch vehicle configurations shown in Table II-1 have adequate performance, but were eliminated from serious consideration because of cost. All of the vehicles had to compete with the basic Atlas-F on an Atlas-F refurbishment cost of $0.9M.

The availability of any replaced Titan II launch vehicle was also investigated; however, none were available due to military test program commitments. The Titan IIIBS (a stretched versions of the Titan IIIB) has good performance capability, but the full recurring cost would have to be paid to obtain this vehicle. Therefore, except for the Titan IIIE/Centaur proof flight opportunity, the Atlas-F/BII configuration offered the most cost-effective selection.

B. EXPERIMENT DESIGN CRITERIA

These criteria define the design, performance, and fabrication requirements for the orbital test module to be launched as a dedicated payload aboard an Atlas-F. These requirements are summarized below:

Orbit Requirements		
Lifetime	-	7 to 14 days
Altitude	-	185 km (100 n mi)
Inclination	-	96°
Launch Dates	-	Consistent with sun-synchronous condition
Low-g Environment	-	10^{-4}g or less
Thermal Control	-	Passive
Attitude Control	-	Maintain Z-local vertical
Power	-	Primary batteries
Communications		
Transmission	-	Tape recorded for replay
Frequency	-	S-Band
Ground Network	-	NASA/STDN

1. Orbit Lifetime

An orbit lifetime of 7-14 days was specified to assure sufficient time between the test expulsion cycles for thermal equilibrium conditions to be established. A time allowance was made for delaying the testing sequence if any unexplained anomalies appeared in the test data. The testing would require only seven days if no interruptions in the normal sequence occurred.

2. Orbit Altitude

The 185 km (100 n mi) orbit altitude selected was consistent with the requirements of 14 days orbit life and the low-g environment. The ballistic number of 22.8 $\frac{W}{C_D A}$ for an orbiting flight test module was high enough for a 14 day orbit lifetime. The atmospheric drag forces on the module would yield an environment of 10^{-6}g; therefore the 10^{-4}g condition was satisfied. Both the orbit lifetime and the drag forces in the module were determined assuming a 1970 Jacchia atmosphere.

3. Orbit Inclination

The launch azimuth for the Atlas-F launched out of WTR was 202.5°. A dogleg in the trajectory reduced the velocity penalty resulting from a westerly launch. This dogleg also yielded a final orbit inclination of 96°, which was compatible with a mission objective to have a sun-synchronous orbit. This sun-synchronous condition

was obtained by constraining the launch dates to a four week interval, centered on either the vernal or autumnal equinox time period. The advantage obtained from orbiting in a sun-synchronous trajectory mode was in the design of the thermal control subsystem and is described in subsection 6 below.

4. Launch Date

Launch dates were fixed in a four-week interval centered on either the vernal or autumnal equinox periods. This constraint placed the launch date in either the Sep-Oct 1973, Mar-Apr 1974, or Sep-Oct 1974 time periods.

5. Low-G Environment

The "g" forces exerted upon orbital test module because of atmospheric drag were determined using a 1970 Jacchia atmosphere density. The analysis indicated the drag forces imposed a value of 10^{-6}g on the flight test module. This is less than the 10^{-4}g used as the experiment design guideline. Therefore, the 185 km (100 n mi) orbit altitude was adequate to generate the required experiment low-g environment.

6. Thermal Control

A passive thermal control was feasible for the orbital test module because of the selection of a sun-synchronous orbital orientation. The orbital conditions described in subsections 3 and 4 above would place the orbital module in continuous sunlight. Also, a Z-local vertical attitude was to be maintained during flight; the sunlight, and shaded surfaces of the flight test module were predictable. Under these conditions, a passive thermal control subsystem was feasible by a judicious use of insulation and/or selection of surface coating. This type of thermal control was used.

7. Attitude Control

The orbital test module was stabilized in a Z-local vertical attitude. The longitudinal axis (X) paralleled the velocity vector of the module as it orbited the Earth. Course attitude control (±5°) was considered adequate to maintain good thermal control and test module communication antenna pointing.

8. Power

The mission duration (7-14 days), and the power consumed while in testing and standby modes, were in a power supply regime best met with primary batteries. A silver oxide/zinc type of primary battery was selected for the module power source.

9. Communications

While in a testing mode, the orbital experiment generated data at a rate of 4000 bps (Ref II-1). These data were stored in tape recorders for later transmission to a ground station on overflight of a designated Spaceflight Tracking and Data Network (STDN) station. Playback speed could be an order of magnitude faster than recording; therefore the 2 to 5 min of ground station contact was adequate time to "dump" the recorder stored data.

Downlink analysis showed that a 2 watt antenna output was adequate for the stabilized S-Band downlink system operating at 185 km (100 n mi) altitude.

The ground swath described by the orbital module was not broad enough to give continuous real-time ground contact with the NASA/STDN. Therefore, the testing data were stored on tape, as described in subsection 9 above. Testing sequence was to be paced by a timer located aboard the module. Overriding of the timer in a contingency requiring a hold could be accomplished by an uplink ground command when in view of a ground network sta on.

C. ORBITAL TEST PLAN

This section describes the orbital test plan proposed for the Cryogenic Flight Test Module when launched as a dedicated payload. Three phases of the test plan are discussed and include the prelaunch, launch and orbital phases.

1. Prelaunch

Final checkout tests will be performed when the orbital flight test module is received at WTR. Visual checks will be made to establish that the module was not damaged in transit. A combined systems test is then performed and on completion of this test the test module is mated to the Atlas-F Burner II launch vehicle and encapsulated in the shroud.

Liquid oxygen loading of the 106 cm (42 in.) storage tank was to be accomplished by using three portable LO_2 dewars with a capacity for 0.569 m^3 (150 gallons) each. The autogenous pressurization sphere was loaded from a cart containing a rack of oxygen k-bottles. After loading, the LO_2 and GO_2 transfer lines would be manually disconnected. The low heat-leaks into the storage tank permit holding on the pad within the time constraints determined by the Atlas launch vehicle itself. Temperature and pressure within the module LO_2 storage tank would be monitored while on the pad.

2. Launch

During the powered phase of flight. no test date would be obtained. The monitoring of temperature and pressure in the module LO_2 storage tank and GO_2 pressurization sphere terminated with liftoff.

3. Orbital

A discussion of the orbital flight test, including the control that was to be maintained over the experiment during the entire orbital mission, is presented in the following section.

The performance and operational characteristics of the dual-screen-liner (DSL) cryogenic storage system are also discussed along with a summary of the anticipated test results from the orbital flight.

a. *Mission Duty Cycle* - The test procedure to be followed during the orbital test is representative of a typical mission duty cycle for an Earth orbiting vehicle such as the Space Shuttle or Space Tug. The operational events to be performed during the mission include: tank fill and hold; boost-to-orbit insertion; low-g coast with venting; autogenous prepressurization; pressurization and liquid expulsion; low-g coast without venting; and liquid outflow to depletion.

As many as 10 separate liquid expulsions will be made to represent ΔV and RCS demands on the acquisition/expulsion device. The total time interval during which liquid is being expelled will correspond to that of a full scale LO_2 tank. The LO_2 expulsion rate will be decreased. The plan specifies that liquid be expelled overboard in a non-propulsive manner.

Pressure control in the storage tank will be of the "minimum pressure" type, where pressure in the storage tank is maintained at the lowest possible level at all times consistent with the NPSP requirements imposed on the system. A ΔP pressure regulator or a pressure switch in conjunction with a solenoid valve will be used to provide the desired pressure control.

A 20 hr portion of a typical 7-day mission duty cycle for a 76 cm (30-in.) dia LO_2 tank is shown in Figure II-1. The simulation was made using the MMC (DSL) Cryogenic Storage Program. The pressures in both the outer vapor annulus and the bulk fluid region are shown, with the pressure difference between the two indicative of the pressure difference maintained across the wetted communication screen. The percentage of LO_2 in the tank at any point in time is also plotted on a zero to 100% scale. A tank input heat leak of 1.58 W/cm^2 (0.5 Btu/hr ft^2) was assumed. A gas annulus gap of 1.9 cm (0.75 in.) and an effective liquid annulus gap of 1.2 cm (0.5-in.) were used. The total initial ullage in the tank was 15.4% with an outer annulus ullage of 14.3%.

Low-g venting, GO_2 prepressurization and also pressurization, and LO_2 expulsion are shown during the 20 hr simulation. The equilibrium pressure condition in the tank following each pressure collapse is at a higher level than that following the previous LO_2 outflow. The venting simulation is representative of a scheme where the pressure is allowed to rise in the outer annulus until a pressure differential is obtained which is slightly less than the pressure retention capability of the communication screen. The vent is then opened and the pressure in the outer annulus is allowed to decrease until the minimum pressure differential needed to support the hydrostatic head of liquid in the low-g environment is reached. The pressure retention capability of the 250 x 1370 communication screen in LO_2 provides for a pressure band of approximately 0.206 N/cm^2 (0.30 psid).

b. *Experiment Control* - Control of the experiment sequence of events will be accomplished by a preprogrammed timer whose action is activated or deactivated by ground command. A timeline of events of a typical mission of the Shuttle Orbiter was used to define the orbital experiment sequence of events. This postulated timeline provides for propellant expulsion of variable amounts at irregular time intervals. This type of simulation of representative flight conditions yields the best appraisal of the operational performance of the DSL passive propellant control device.

A digital program timer made by the Data Science Corporation in San Diego, California, would be adequate for this test. This timer was successfully used on the Martin Balloon Launched Deceleration Test (BLDT) program. The timer can accommodate 10 preprogrammed events. A typical Shuttle Orbiter timeline can contain as many as 15 discrete ΔV events. This number will be selectively reduced to 10 for the test timeline. (Two timers could be used in series if more than 10 events are considered necessary.)

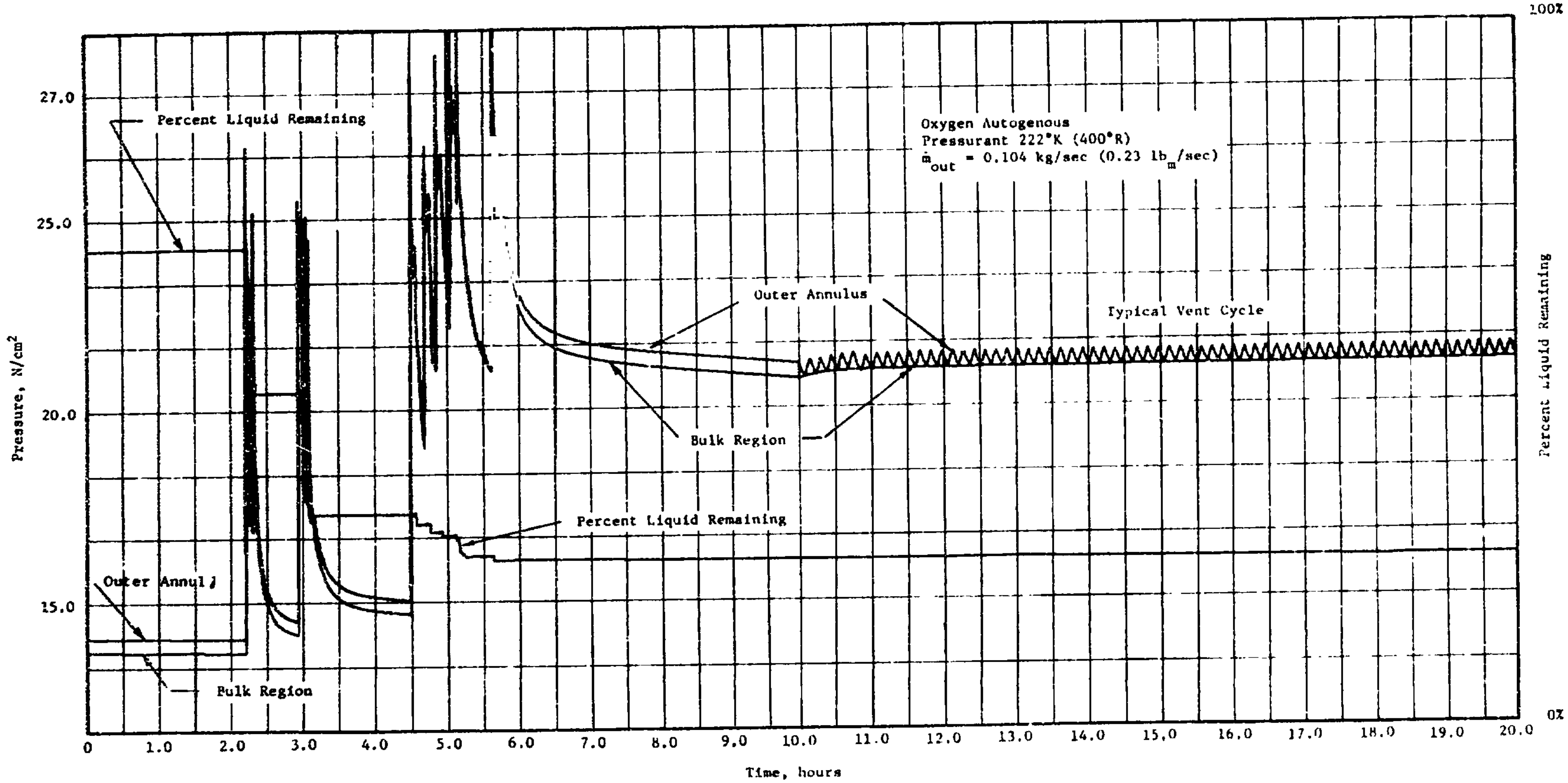

Fig. II-1 20 HOUR SIMULATION OF A TYPICAL MISSION DUTY CYCLE

Ten minutes of continuous data gathering are proposed during periods of transient conditions, e.g., pressurization, expulsion of LO_2, and venting. Since this 10 min interval is of longer duration than the time period the orbital module is in continuous contact with any ground station, the test data will be recorded on a tape recorder. On ground command, the tape recorder will playback to a ground station. This playback command will be made at the earliest possible opportunity.

Recording time for the tape recorder will be in the range of 10 to 20 min. One test is planned where the recorder will run approximately 96 min - the full recording capacity of a Leach MTR-2000 recorder. During this long duration, observation of temperature and pressure transients will be made. They will be measured from initial prepressurization of the tank, prior to expulsion of a quantity of LO_2, through the expulsion period and subsequent tank pressure collapse. The MMC DSL program simulation indicates that these transients will take about 90 min.

c. System Performance - Throughout the orbital test, data will be obtained to verify that the DSL capillary acquisition device and the screen liner in the propellant feedline control the bulk liquid oxygen by keeping it from the walls during low-g storage. The DSL also stabilizes and controls the liquid in the flow channels to assure gas-free liquid expulsion. Control of the bulk liquid is necessary because it tends to minimize stratification effects, if any, and reduces propellant vaporization and resultant vented mass. In addition, data should verify that vaporization in the controlled liquid region does not result during venting.

An additional stability consideration with cryogenic storage is the ability of the screen and perforated plate material to remain wetted under the imposed thermal environment. Heat leak enters the system through the tank wall and supporting structure of the capillary assembly, through soakback from the propellant feedline, and by the warm pressurization gas. Stability loss due to screen dryout is critical to the efficient operation of the system. Of particular importance is the ability of the communication screen to rewet following any dryout, thus providing the required pressure support to maintain the bulk liquid inside the screen compartment.

A measure of the system performance to supply gas-free liquid at the exit of the feedline is the expulsion efficiency. This efficiency depends on the ability of the screen forming the liquid annulus to prevent pressurization gas and propellant vapor from entering the liquid annulus and being expelled. During expulsion,

the bulk liquid is reduced and the screen area exposed to liquid flow from the bulk volume to the liquid channels is decreased. As this occurs, the pressure drop for flow through the screen increases due to the increased flowrate per unit area. The DSL design will permit nearly all of the bulk liquid to be depleted before gas in ingested into the liquid annulus and expelled. Thus, the volume within the liquid channels is unavailable propellant. It is therefore, desirable to design the smallest possible liquid annulus while still satisfying the liquid flowrate requirements and flow losses.

d. *Operational Characteristics* - A matrix of operational characteristics for the DSL acquisition/expulsion system is presented in Table II-2. The marks indicate those parameters that are critical to the performance during the orbital flight. High-g, as well as low-g, performance is considered with the multiple sequence section indicative of system operation during the major portion of the flight. The initial low-g operation differs from the multiple sequence only in that capillary retention in the feedline is not required.

e. *Test Results* - The data obtained during the orbital experiment will be used to verify system performance and identify system operational characteristics. More specifically, the data will be used to verify that the system provides gas-free liquid expulsion at the required flowrates and maintains tank pressure control under the low-g operating conditions without loss of liquid from the system. Tank pressure can be controlled by efficient venting of saturated or superheated vapor. If the acquisition/expulsion system provides adequate fluid stability during the imposed acceleration environment, the liquid will not become positioned over the vent and will not be vented when tank pressure relief becomes necessary. In addition, the data should verify that vaporization in the controlled liquid region does not occur during venting. During pressurization of the storage tank, pressure data will indicate dryout and rewetting of the communication screen. Pressurization with autogenous pressurant should also tend to collapse any bubbles that may have formed in the liquid channels due to any superheating of the liquid during venting.

When data reduction of the orbital flight data has been accomplished, correlation will be made with analytical predictions from the DSL Cryogenic Storage Program. This program can simulate pressurization, venting and liquid draining for a complete mission duty cycle.

Table II-2 System Operational Characteristics

Test Performance Function		Capillary Retention-Storage Tank	Capillary Retention - Feedline	Vapor Entrapment	Compression of Entrapped Gas	Communication Screen Operation	Vapor Breakthrough-Liquid Channels	Vapor Breakthrough-Bulk Region	Liquid Dropout	Vapor Bubble Growth	Vapor Bubble Collapse	Wicking (Rewetting)	Heat Sock Back	Slosh/Propellant Dynamics	Start-up & Shut-Down Transients
One-g	Tank Fill	X		X											
	Pad Hold	X		X						X			X		
High-g Boost	Launch	X		X			X		X					X	X
Low-g	Establish Vapor Region in outer Annulus	X		X	X							X			X
	Coast with Venting	X							X	X			X		
	Prepressurization	X			X	X		X			X	X			X
	Pressurization	X			X	X		X			X	X			
	Liquid Outflow	X	X											X	X
	Pressure Collapse	X	X					X	X			X			X
Low-g, Multiple Sequence	Coast with Venting	X	X						X	X			X		
	Prepressurization	X	X		X	X		X			X	X			X
	Pressurization	X	X		X	X		X			X	X			
	Liquid Outflow	X	X											X	X
	Pressure Collapse	X	X					X	X			X			X
	Coast without Venting	X	X			X	X	X				X	X		
Low-g	Liquid Outflow to Depletion	X	X			X	X	X							X

Correlations that will be made with the analytical predictions include: 1) pressure rise in the tank during coast; 2) pressure response during intermittent vapor venting; 3) liquid temperatures in the controlled liquid regions during coast; 4) liquid temperatures in the bulk region during the thermal transients associated with autogenous pressurization; and 5) pressure decay in the outer annulus and bulk storage regions following pressurization and outflow.

D. TEST ARTICLE ANALYSIS

This section describes the analysis of the flight test article and the supporting subsystems making up the complete flight test module. Candidate subsystem hardware was evaluated in order to establish a good total system design.

1. Communications Subsystem

The communications sybsystem requirements will be reviewed so as to analyze, select, and define the subsystem's configuration, characteristics, performance, and interfaces. Scientific and engineering data requirements will be identified as a function of time from the sequence of events established in the Flight Test Plan. Alternative methods of storing, processing, sequencing, formatting, and transferring data between the orbital flight test article and the ground stations will be considered.

a. *Candidate Communications Subsystems* - Two candidate communications subsystems designs are shown in Figures II-2 and II-3. These two systems are compatible with the STDN, as described in the STDN User's Guide (STDN 101.1 April 1972). The two figures show candidate configurations for a nonstabilized orbital experiment module. They provide omnidirectional antenna patterns and two transmitters operating at about 50 MHz difference in frequency each, feed the two antennas.

The use of an attitude control subsystem on the experiment module simplifies the communication subsystem design by eliminating the need for omnidirectional antenna and the dual transmitter equipment. A single transmitter is required.

The design in Figure II-3 has the advantage of the RF components being similar to those utilized by Martin Marietta for the BLDT Program.

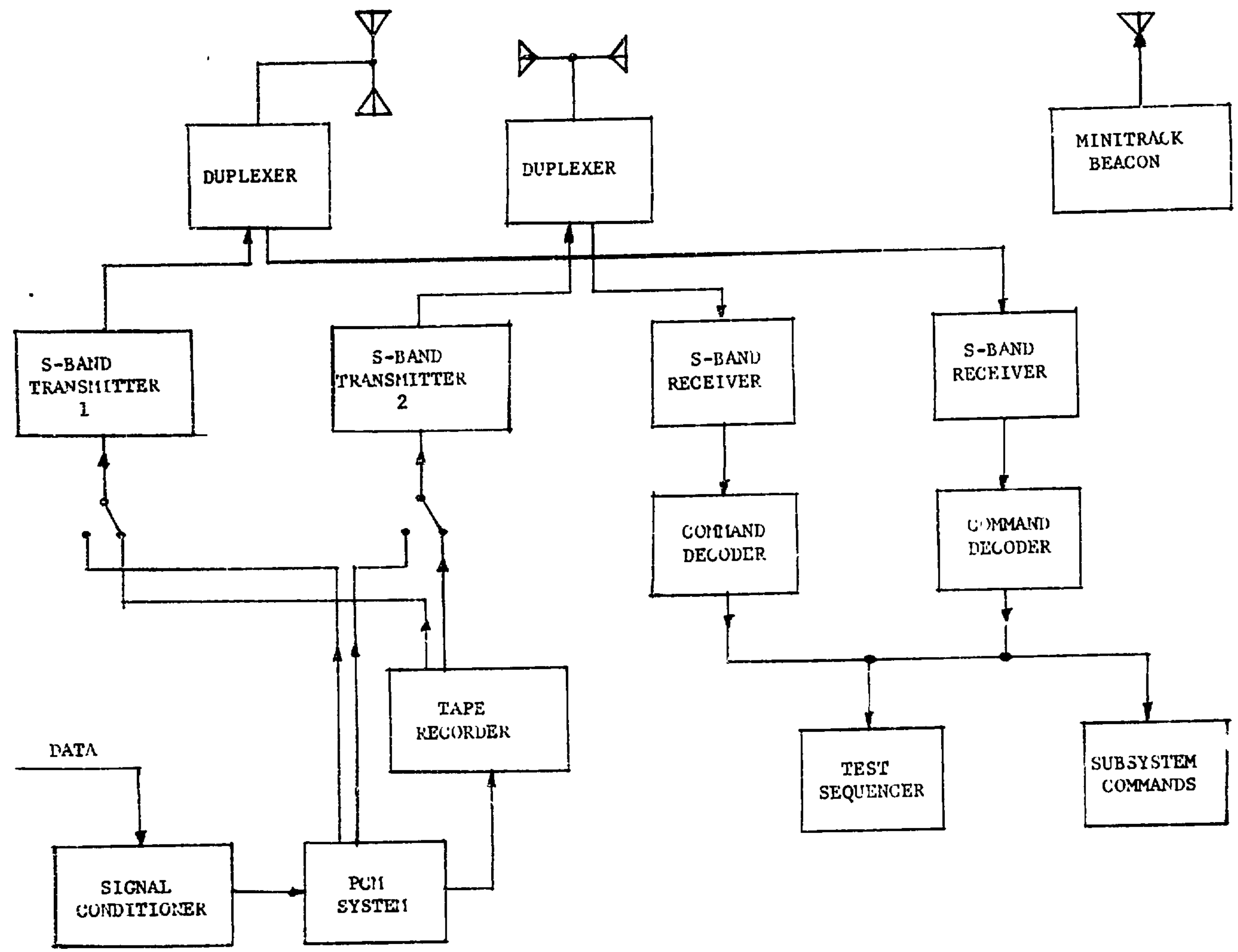

Fig. II-2 S-Band/S-Band Candidate Data and Communication Subsystem

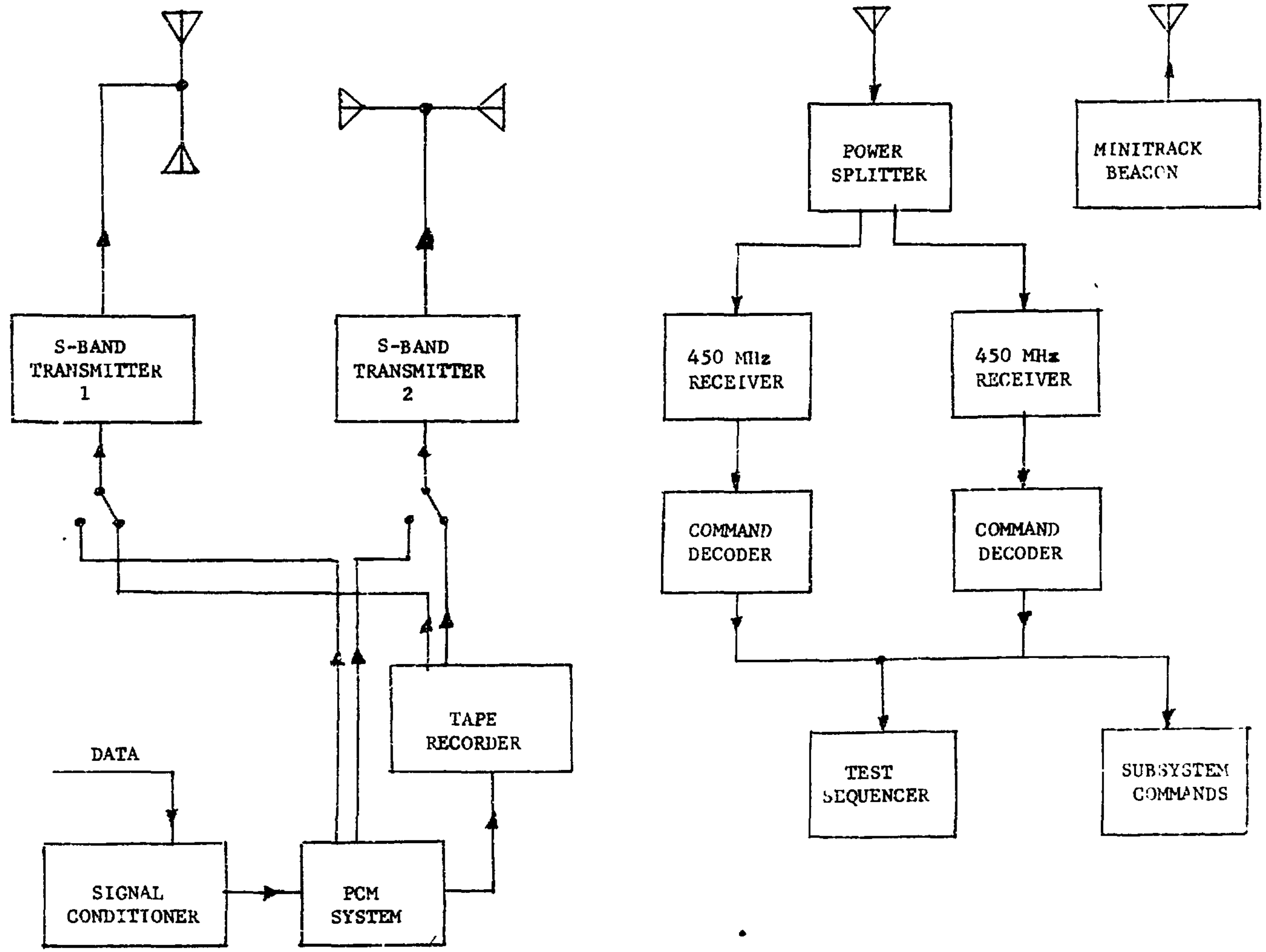

Fig. II-3 S-Band/450 MHz Candidate Data and Communication Subsystem

Figure II-4 shows the flow of activities for performing the data and communications tasks outlined in this subsystem program plan.

The Support Instrumentation Requirements Document for Goddard Space Flight Center will be generated. A similar document will also be prepared for Vandenberg Air Force Base.

Specific frequency authorizations will be obtained prior to procuring the telemetry transmitters and command receivers. The environmental levels for the test vehicle will be determined prior to procuring this equipment.

Preliminary antenna tests are also planned prior to procuring the antenna.

b. Data Acquisition and Reduction - Data acquisition and reduction will be covered in the Support Instrumentation Requirements Document to Goddard.

The major requirement is provision of real-time and near real-time information to a Project Operations Control Center at Goddard. The orbital operations will be conducted from the Project Operations Control Center.

c. Procedures for STDN Support from GSFC - A Support Instrumentation Requirements Document will be prepared as per directions in Section 2 of STDN 101.1 (Ref II-5). This input is shown in preliminary form in the following listed items:

1) Program Purpose - To demonstrate the capability of a capillary screen device to passively control liquid cryogenic propellants during extended periods of low-g environments. Liquid oxygen will be used in the orbital test article.

2) Launch Dates - Near-vernal or autumnal equinox periods in 1974 or 1975 if a sun-synchronous orbit is selected.

3) Duration - Maximum of 14 days, limited by battery life and orbit lifetime. Nominal test period planned as 7 days.

4) Required Support - Minimum of one real-time pass to start a programmed test sequence and one pass to play back recorder tape.

5) Flight Events - The vehicle will be placed in orbit via USAF/SAMSO SCF stations prior to monitoring by STDN operations. The permissible module tumble rate will be limited to a maximum value if orbited in an unstabilized mode. The telemetry transmitters will be ground commandable.

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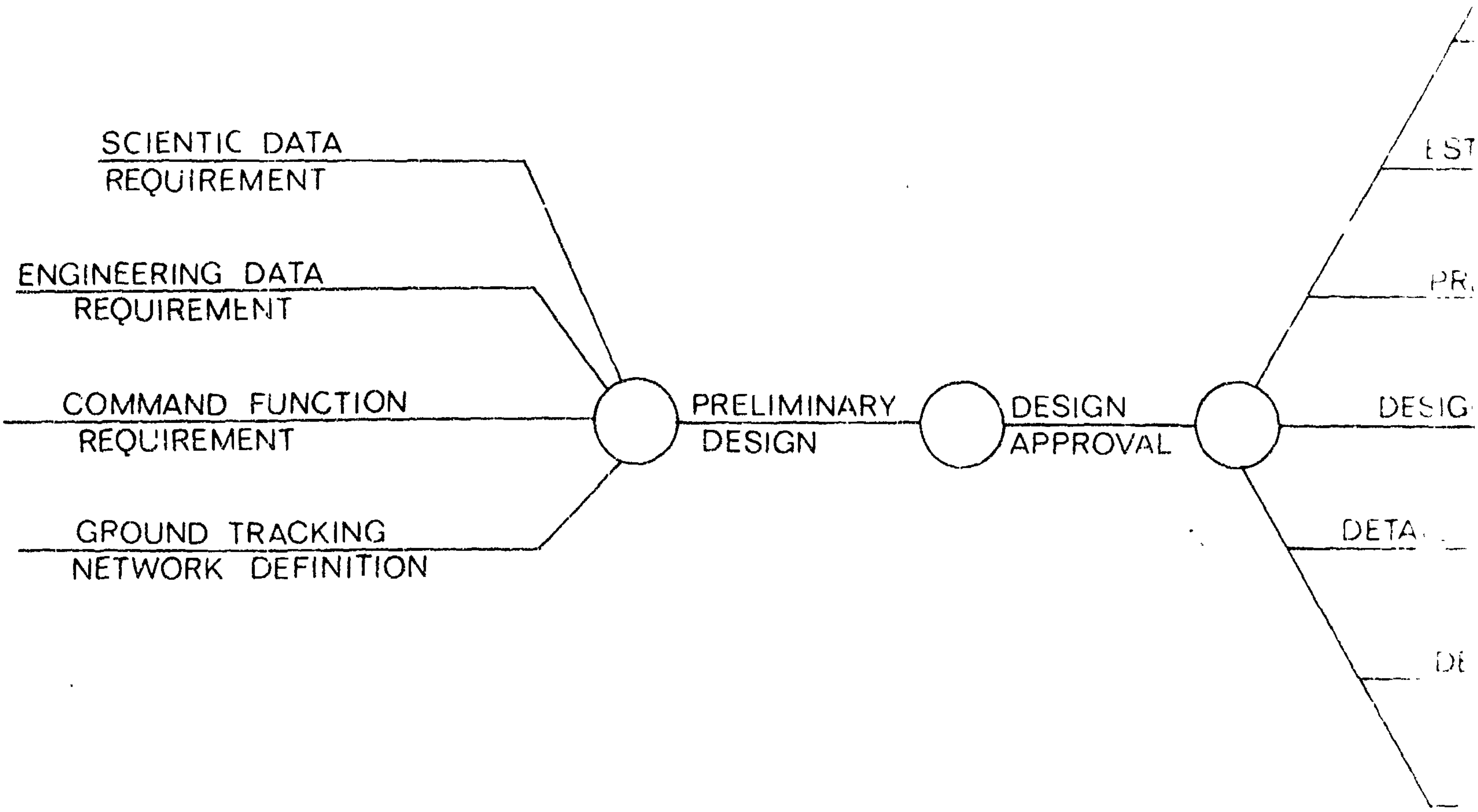

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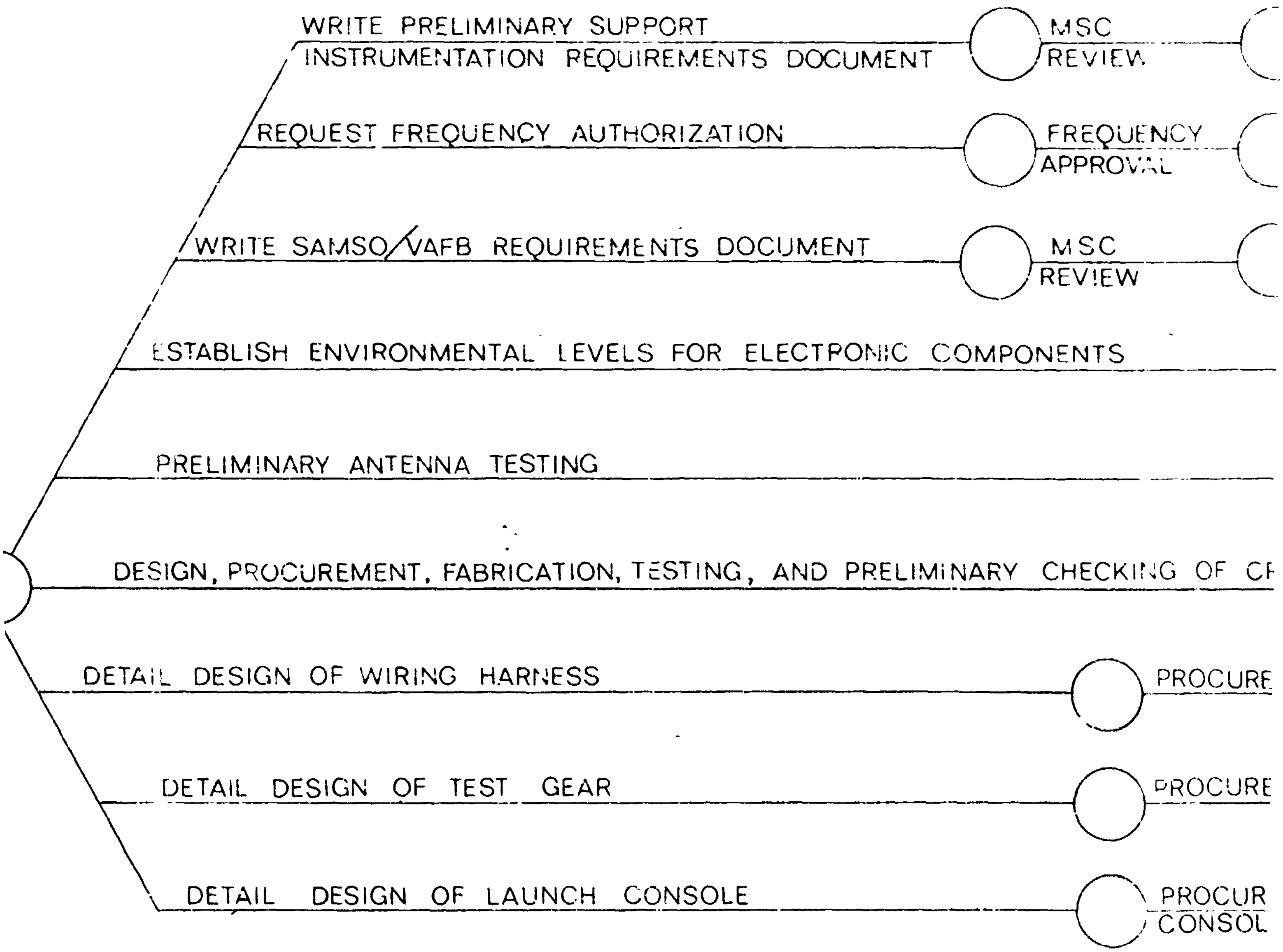

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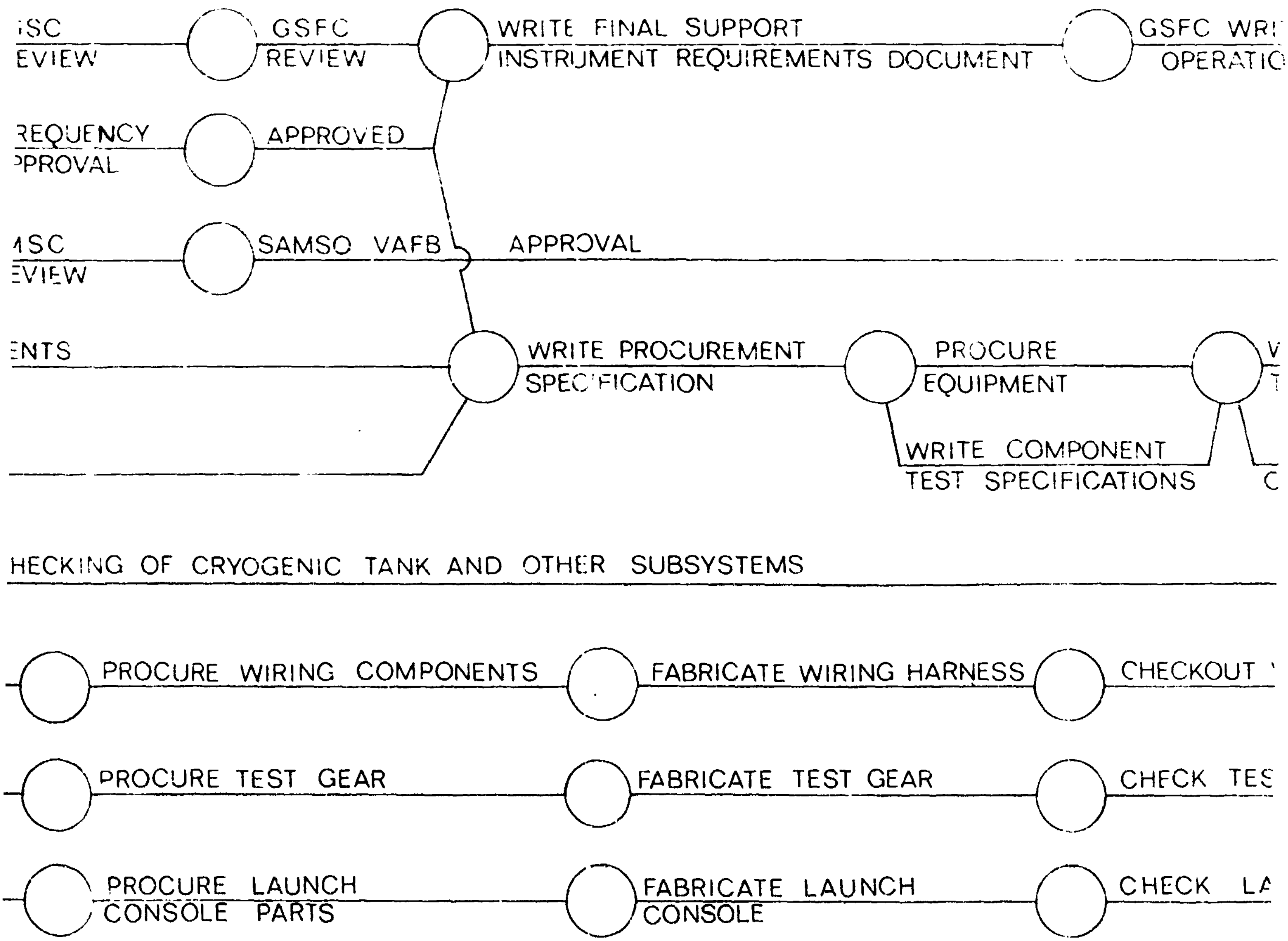

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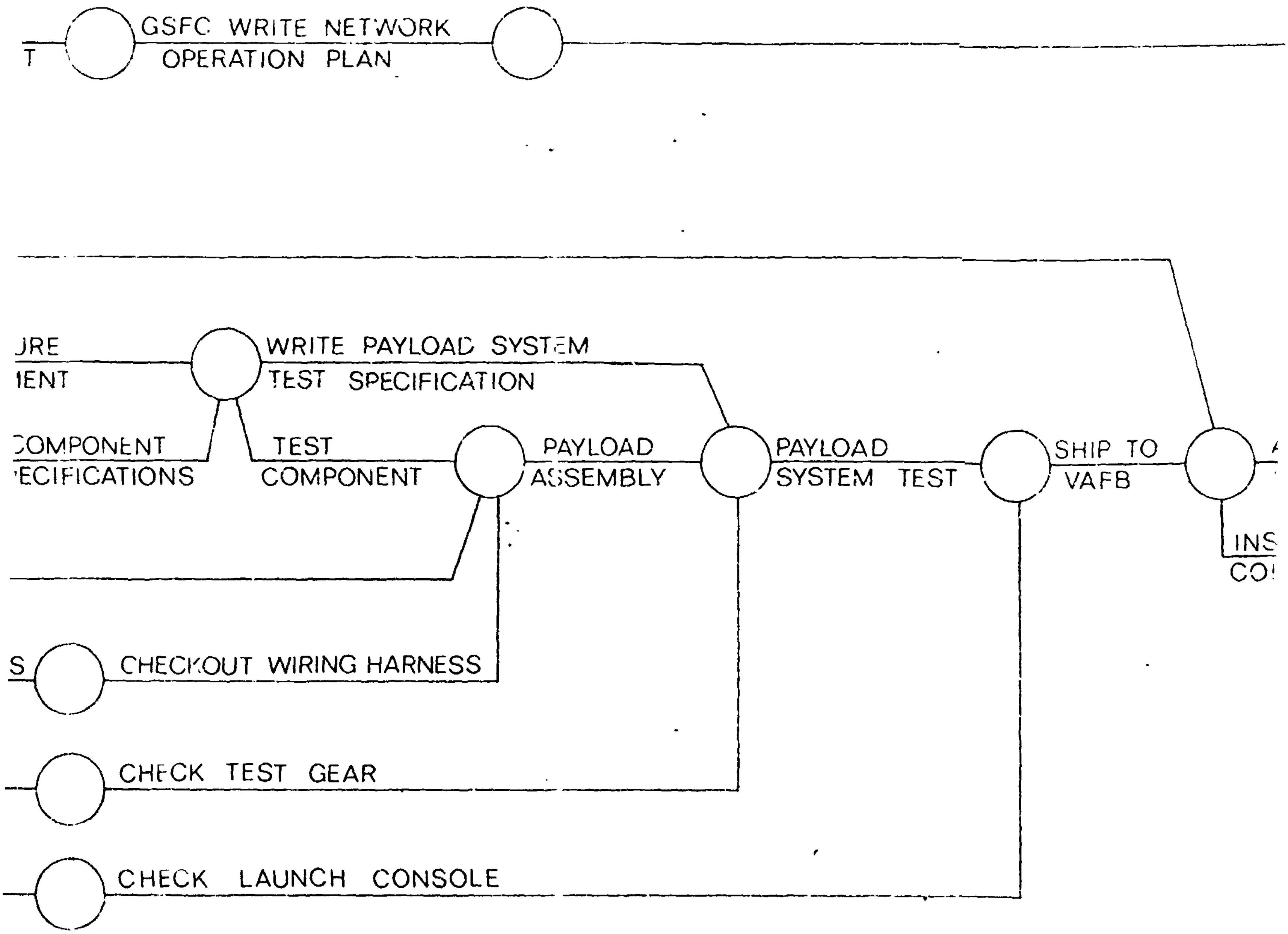

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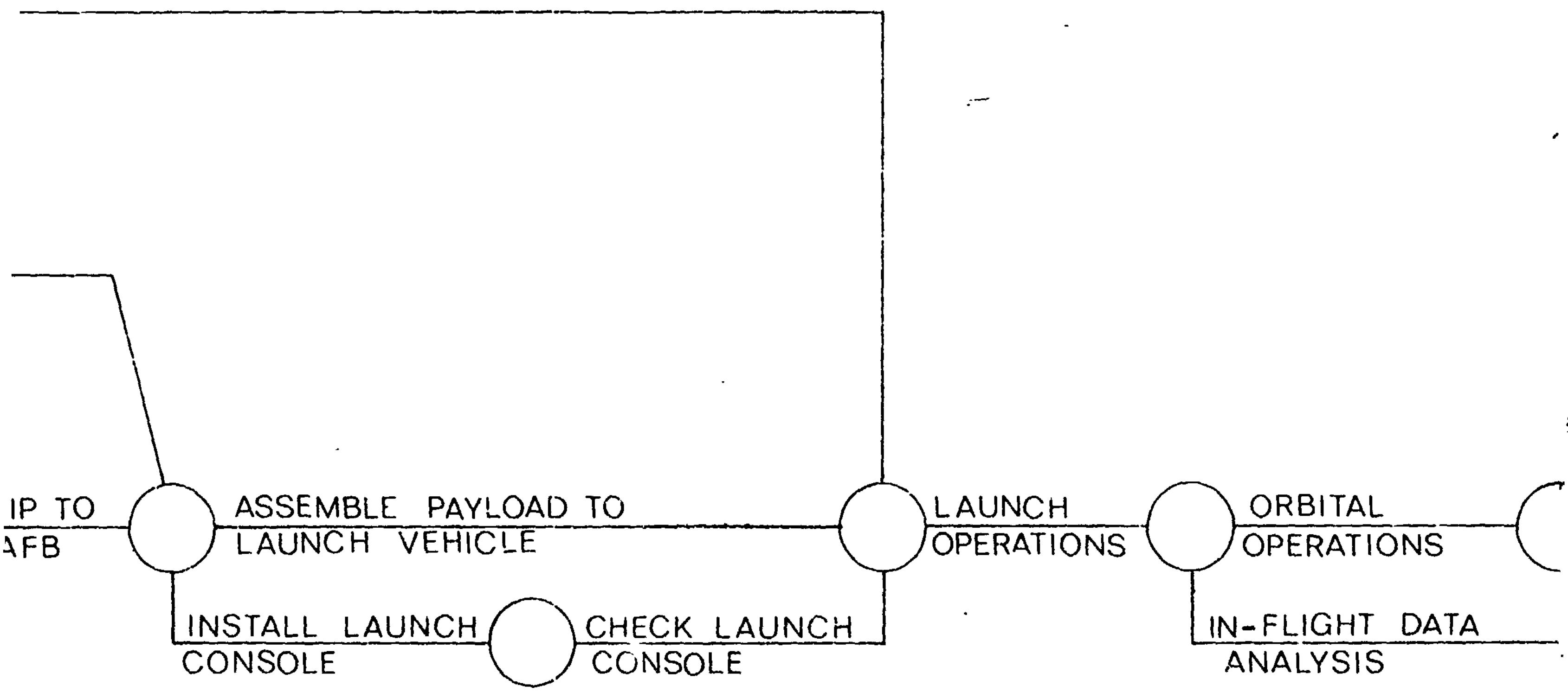

Fig. II-4 Du

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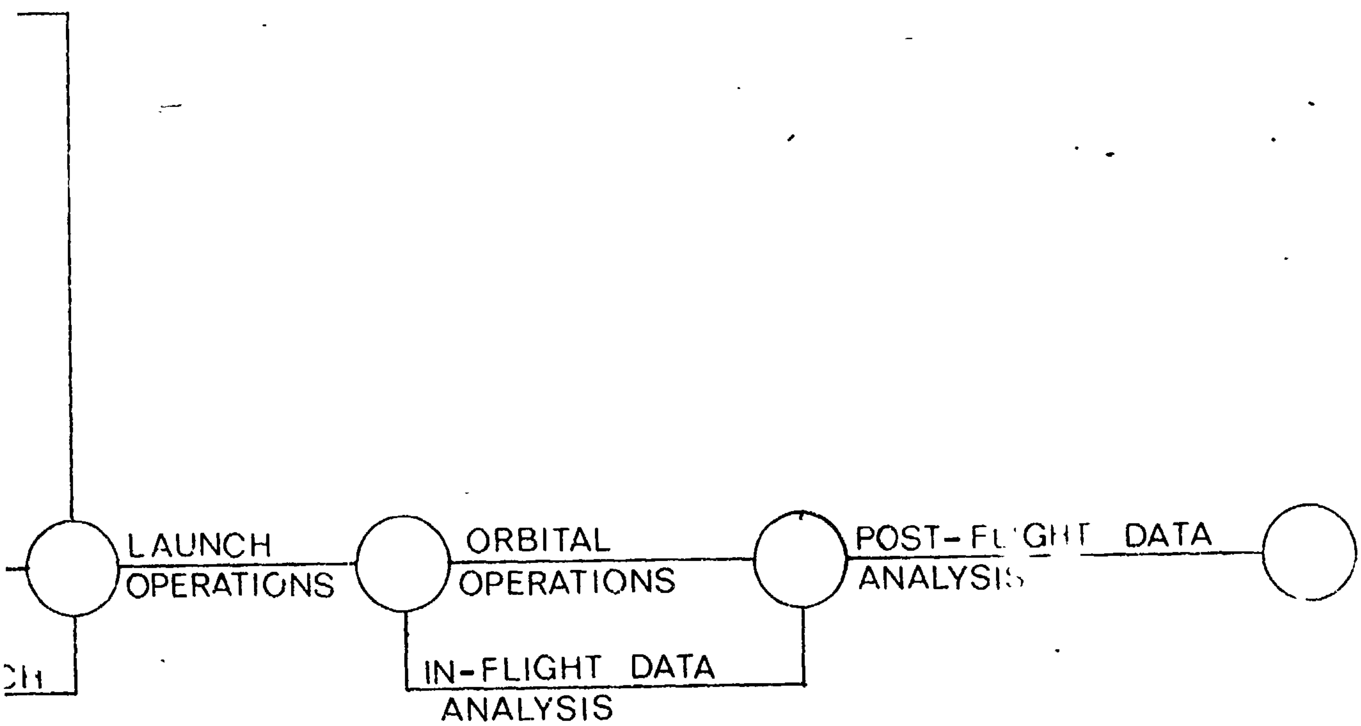


Fig. II-4 Data and Communication System Work Flow

6) Telemetry and Tracking Systems - Two S-Band telemetry transmitters will transmit the same PCM format to provide approximately omnidirectional coverage for the orbital experiment module flown in an unstabilized mode. With the inclusion of an attitude stabilization in the configuration concept, only one S-Band telemetry transmitter is required and it transmits on a single antenna pointing approximately to nadir.

   Two 450 MH_z command receivers will receive ground commands utilizing the Interrange Instrumentation Group (IRIG) tone system.

   A 136 MH_z VHF tracking beacon will provide tracking information to Minitrack Network.

7) Telemetry Parameters - Telemetry parameters will be provided at a later date. Real-time data will be approximately 4 kbps if the philosophy of transmitting partial real-time test data to a ground station, before sequencing the next test event, is used. Tape recorder playback data of approximately 20 kbps (possibly 72 kbps depending on recorder choice) are adequate.

8) Recorded Data - Magnetic tape of each station pass is to include timing, project operations voice net, both telemetry links, and the commands sent to the orbital experiment module.

9) Ground Communications - Two voice links between Project Operations Control Center at Goddard and Payload Launch Control Console at Vandenberg Air Force Base are required; also two teletype circuits to transfer orbit injection vector from Vandenberg to Goddard.

10) Unusual Requirements - Real-time command control to the Project Operations Control Center is the only item.

11) Final Data Availability - December 1, 1973, is baselined.

2. Power Subsystem

An electric power profile based upon the duty cycle will be updated to include any additional power requirements identified in the subsystem analysis. Based upon these power profiles, the required battery power system (AgZn battery) will be sized.

3. Environmental Control Subsystem

Although environmental control is frequently used synonymously with thermal control, other forces presenting a threat to the experiment module viz., sonic and vibrational forces during launch, and micrometeoroid impact during orbit, will be investigated.

a. *Thermal Control* - The objective of this task is to define a simple, reliable, thermal control system design meeting the requirements of the orbital module equipment. A passive system is proposed, and if a sun-synchronous orbital condition and a stabilized experiment module is chosen, this type of thermal control is practical.

Acceptable temperature limits will be established for the orbital module subsystem elements. The task will generate an equipment list defining acceptable temperature limits. The list will be used in the selection of the thermal control method to be used. The response of critical equipment to the imposed thermal conditions will identify the methods of thermal control that best meet the requirements. If heat is required, heat retention will be maximized to minimize need for additional heat sources; if heat is excessive, dissipation will be maximized. Heat transfer rates will be defined for the critical equipment and time histories of electrical power required, if any, for thermal control will be compiled. These time histories will be used to determine the overall electrical load profile. Based on the heat transfer rates, a thermal control subsystem approach will be selected and defined. Analyses will be of sufficient depth to ensure feasibility, compatibility with other subsystems, adequate performance and reliability levels, and to define basic physical characteristics and interfaces.

4. Attitude Control Subsystem

The stabilization requirements are of the order of ±5° in all three axes. This is a coarse ACS requirement and is readily met with existing candidate elements. Each of the candidate concepts will be analyzed for their performance, cost, weight, power, and reliability characteristics.

Where existing subsystems with greater accuracy than required are available, cost will be the principal criteria for selection.

5. Structure Subsystem

The objectives of this task are to develop an efficient orbital experiment module structure to integrate the subscale LO_2 experi-

mental tank into its carrier, and to structurally support the necessary other subsystems of the orbital experiment.

The structural criteria for payloads mounted on the Atlas-F at booster Station 502 will be determined. The environment of the payload during Atlas-F thrust build-up and flight, in terms of aerodynamic loads, longitudinal acceleration, acoustic noise, and launch vehicle propellant sloshing will be determined.

6. Instrumentation Subsystem

The objective of this task is to determine and identify the number of the orbital experiment module status points to be telemetered.

The scientific instrumentation will be designed to provide the data necessary for analyzing the performance of the flight test article. Primarily, instrumentation must be provided to show that only vapor free liquid is delivered through the propellant feedline and only liquid-free vapor is vented. Secondarily, instrumentation will be included to assess the condition of the flight test article during the test. Table II-3 summarizes the instrumentation proposed for the flight test article. Figure II-5 presents a preliminary layout of the sensors in the flight test article.

Temperature measurements will be made with platinum resistance sensors. Rosemount Engineering Co. Model 146MA sensor meets the requirements of the orbital experiment and will be considered as a supplier. These sensors are small and can be located in the test article to accurately measure temperatures at critical locations, e.g., liquid outflow temperatures, temperature differentials across screens, and vent gas temperatures.

It is proposed that liquid/vapor conditions be detected using constant resistance wire sensors. A unit suitable for this application is manufactured by Sundstrand Data Control, Model 2641. This unit is accurate to 0.15 cm (0.06 in.) of liquid level with a response time of 0.15 sec when passing from gas to liquid. It will withstand a 50 g sinusoidal vibration to 2000 Hz. The output of the unit can be conditioned using a standard unit to produce a signal compatible with the airborne telemetry system. These sensors have adequate sensitivity to confirm the quality of the flow in both the vent and outflow lines. This high sensitivity also makes it possible to evaluate the fluid behavior in the propellant tank.

Table II-3 Flight Test Article Instrumentation

Sensor Location/Type	No.	Range N/cm^2	psia	Accuracy
Pressurization Sphere Pressure	2	0-2410	0-3500	±0.5%
Tank Bulk Region Ullage Pressure	1	0-35	0-50	±0.5%
Tank Vapor Annulus Pressure	1	0-35	0-50	±0.5%
Outflow Line Pressure	1	0-35	0-50	±0.5%
Screen Differential Pressure	1	0-0.35	0-0.5*	±1.0%
		°K	°R	
Tank Temperature	32	83-278	150-500	±0.056°K(±0.1°R)
Pressurization Sphere Temperature	2	83-278	150-500	±0.056°K(±0.1°R)
Vent Gas Temperature	1	83-278	150-500	±0.056°K(±0.1°R)
Outflow Line Temperature	2	83-278	150-500	±0.056°K(±0.1°R)
Tank Liquid/Vapor Sensors	19			±0.15 cm(±0.06 in.)
Vent Line Liquid/Vapor Sensor	1			±0.15 cm(±0.06 in.)
Outflow Line Liquid/Vapor Sensor	2			±0.15 cm(±0.06 in.)
Outflow Line Flowmeter	1			±0.5%

*psid

Pressure measurements will be made using strain gage transducers such as Taber Teledyne Company Series 2210 pressure transducer. The environment of the flight test article tank makes it desirable for the transducers to be located outside of the tank and connected to the sensing points by tubing. The pressure transducer for the tank and outflow line provide a sufficient redundancy to preclude the need for backup sensor. However, two transducers have been provided in the pressurant sphere to improve the probability that this data is returned.

A flowmeter will be installed in the outflow line. It is proposed to use a pitot tube type flowmeter as this will not only provide flow rate data but can also provide rough backup data to the liquid/vapor sensors if two phase flow occurs in the line. Flowmeters of this type are manufactured by Ellison Instrument Division of Diete-

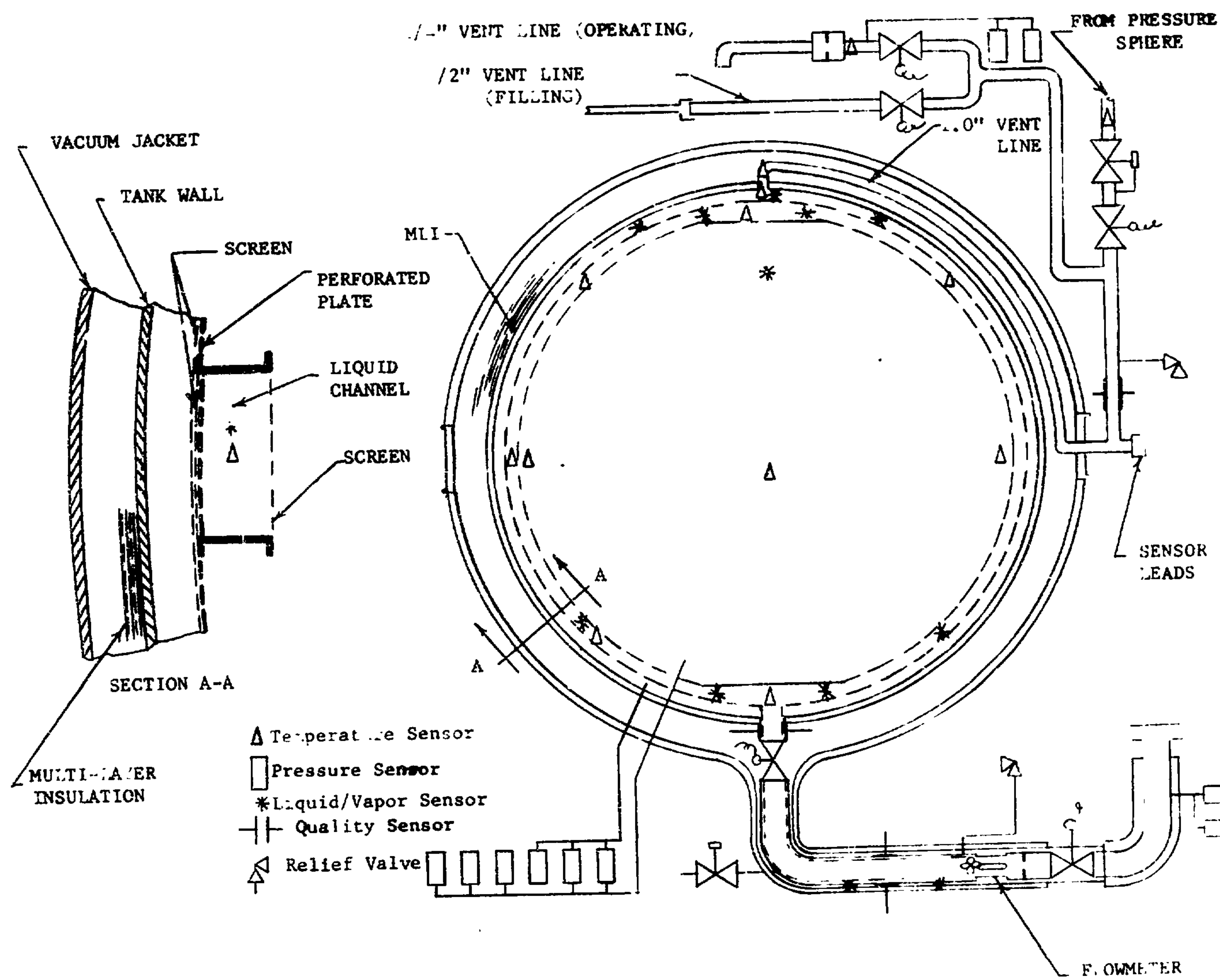


Fig. II-5 Flight Test Article Instrumentation

rich Standard Corporation. Their type 710 Annubar flowmeters have been tentatively selected for use in the flight test article.

The engineering instrumentation will provide general information on the orbital module status. Such items as environmental conditions, module attitude, power supply conditions, and telemetry conditions will be monitored. The exact requirements for this instrumentation cannot be defined until the launch vehicle is specified. When the launch vehicle is specified it will be necessary to evaluate the engineering instrumentation requirements and coordinate these requirements with those of the flight test article.

E. TEST ARTICLE DESIGN

The design of the flight test article and its supporting subsystems is described in this section.

1. Flight Test Article Design

a. *Guidelines* - The design of the orbital flight test article is based upon the following general guidelines:

1) The mission duration will be from 7 to 14 days.
2) The system to be flown will be a low pressure <34.3 N/cm^2 (50 psia) liquid oxygen DSL acquisition/expulsion device.
3) Pressurization will be accomplished with warm $>222°K$ (400°R) autogenous pressurant from a high pressure gas storage container.
4) The acceleration environment during the low-g portion of the flight should not exceed 10^{-4}g.
5) Complex handling and ground fill procedures will be avoided as well as other ground servicing requirements which complicate prelaunch servicing.

b. *Flight Test Article Description* - The subscale liquid oxygen flight test article consists of a spherical storage tank with a screen liner acquisition/expulsion device and a feedline, complete with a screen liner to hold liquid away from the wall for immediate supply to satisfy liquid outflow requirements. The storage tank and feedline are constructed of 300-series stainless steel. Both the storage tank and feedline are insulated with high performance multilayer insulation, e.g., aluminized mylar with nylon net spacing, and are enclosed in aluminum vacuum-jacketed shrouds.

Provision is made for LO_2 fill and drain, storage tank venting and autogenous pressurization, vacuum jacket pump down and pressure relief, LO_2 outflow, simultaneous GO_2 venting of the storage tank and feedline, supply of warm autogenous pressurant from a high pressure supply vessel and instrumentation needed to obtain data for verification of system performance and operational characteristics. A flight test article schematic is shown in Figure II-6. All of the valves, meters, and instruments are to be flight qualified to assure orbital flight reliability. In addition, all tank materials, valve seats, pressure transducers, temperature sensors, etc., must be liquid oxygen compatible.

c. *Storage Tank* - The storage tank size has been reduced to two options by the selection of an Atlas-F/Burner II as the baseline launch vehicle. A 76 cm (30-in.) dia tank containing approximately 227-kg (500-lb_m) LO_2 can be placed in a 185 km (100 n mi) Earth orbit using A Burner II module with no solid motor as the carrier. A 106 cm (42-in.) dia tank containing 680 kg (1500 lb_m) LO_2 can be placed in a 185 km (100 n mi) orbit if a solid motor is used in conjunction with the Burner II module. The acquisition/expulsion system for these two storage tank sizes is identical in design with differences in dimensions producing differences in initial ullage volume and volumetric loading efficiency. A layout drawing of the storage tank with a screen liner retention/expulsion device enclosed is shown in Figure II-7. The propellant control device consists of a complete spherical liner of 250 x 1370 Dutch Twill stainless steel screen separated by an annular region from the tank wall. It has been shown that liquid-free vapor venting of this device is dependent on the gap thickness of the outer vapor annulus, which tends to fill with vapor from evaporation of liquid at the screen surface due to tank heat leak. Vent rate and vent frequency are functions of this gap size.

A vapor-free liquid reservoir is formed by 12 separate liquid flow channels attached to the outer screen liner and joined in a manifold arrangement over the tank outlet. These flow channels are formed from 325 x 2300 Dutch Twill screen. Both the outer screen liner and the outer screen of the liquid flow channel are supported by perforated plate, as shown in Section A-A of Figure II-7.

The LO_2 bulk storage region is located within the outer screen. The liquid flow channels provide a continuous liquid path from the bulk liquid region to the tank outlet. The channels also supply liquid to the outer screen where vaporization is preventing the tank heat leak from reaching the bulk liquid.

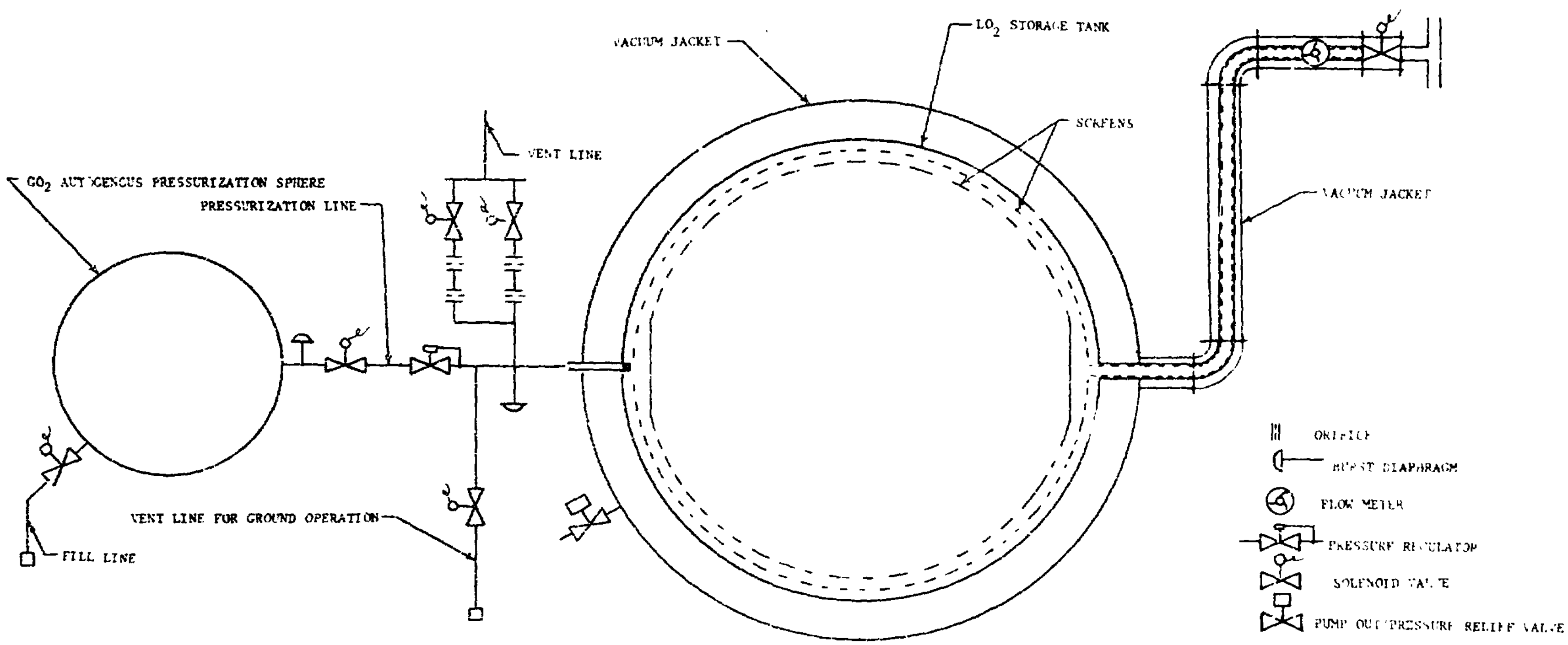


Fig. II-6 Flight Test Article Schematic

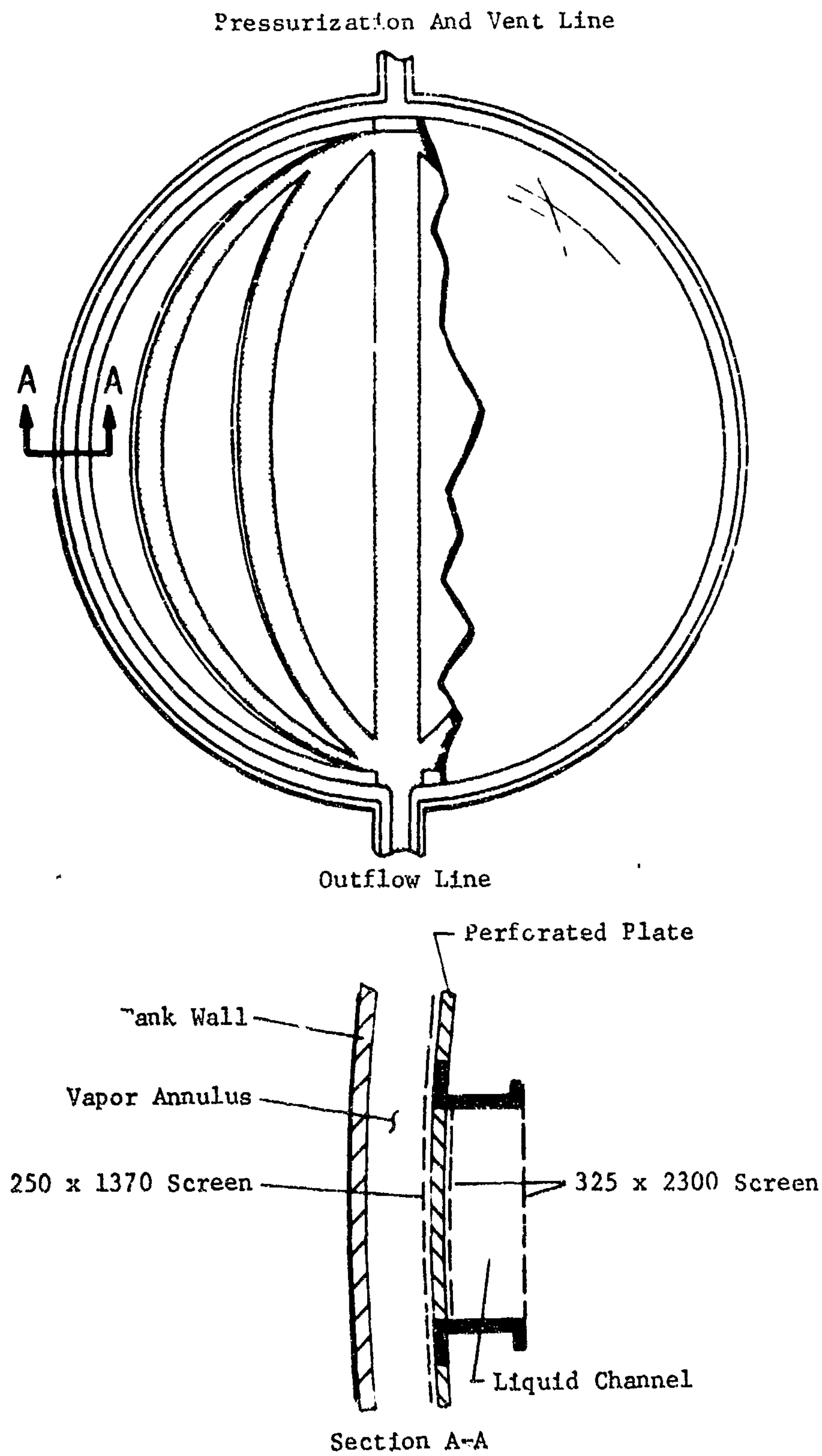


Fig. II-7 Typical Capillary Screen Assembly

The outer vapor annulus and bulk fluid region are passively connected by maintaining the 250 x 1370 screen wetted for regulation of the pressure differential between the two regions. This pressure differential, which is maintained greater in the vapor annulus, must support the hydrostatic head of the bulk liquid under the imposed acceleration environment. During the orbital test, where there is little or no hydrostatic head, the pressure in the gas annulus must not fall below the inner-ullage pressure. Conversely, wher the pressure in the gas annulus rises above the bubble point of the 250 x 1370 screen, gas will preferentially enter the bulk fluid region rather than the liquid region formed by the flow channels, since the bubble point of the 325 x 2300 screen is greater. Pressurization gas is introduced into the outer vapor annulus and is likewise communicated through the 250 x 1370 screen, preventing ingestion of gas bubbles into the controlled liquid flow channels.

The storage tank is insulated with multilayer insulation (MLI) composed of 20 layers of aluminized Mylar with each layer separated by nylon netting. A combined vacuum pumpout/pressure relief valve is incorporated in the vacuum jacket.

Beech Aircraft Corporation, Boulder, Colorado, critiqued the LO_2 storage tank and feedline design proposed for the 76 cm (30 in.) dia flight test article (Ref II-7). Emphasis was placed upon the particular features of: ease of fabrication, support system, cleaning, inspection, handling and maintenance, and tank loading. Their evaluation showed the proposed design to adequately meet these design objectives.

d. Feedline - The recommended feedline configuration to provide sub-cooled liquid at an RCS pump or OMS interface is a screen liner positioned near the feedline wall with vapor vented from the space between the liner and wall, as required, to control system pressure. The outer vapor annulus surrounding the liquid core in the feedline is open to the vapor annulus in the storage tank. A single vent control system is adequate to provide the required pressure control of the storage tank/feedline delivery system. A 325 x 2300 Dutch twill screen liner is positioned inside a stainless steel feedline insulated with MLI and enclosed in an aluminum vacuum jacket. The feedline will have a minimum of two bends. Liquid that is expelled through the feedline is dumped overboard in a non-propulsive manner.

In addition to the storage tank and feedline assembly, pressurization and vent control systems are provided along with the connecting lines and valving. A 1380 N/cm^2 (2000 psi) GO_2 pressurization sphere supplied warm pressurant >222°K (>400°F) to a regulator connected in

series with a solenoid valve. The vent control system is designed to handle both the pad-hold venting where the vented gaseous oxygen must be expelled overboard outside the payload fairing, and the venting of vapor from the outer annulus region of the tank during the low-g orbital flight. The latter system may employ several viscojets in series, with vent flow initiated by a pressure switch connected to a solenoid valve.

e. *Instrumentation* - The number and location of sensors in the spherical LO_2 tank and associated feedline are shown in Figure II-5.

Pressures and temperatures in the various regions of the tank will be recorded during the flight test. The liquid level sensors will be monitored to determine the presence of liquid or vapor at a particular position in the device during the duration of the test. Platinum resistance temperature sensors are used to verify liquid-free vapor in the gas vent and vapor-free liquid in the LO_2 flow channels and the outflow line. Storage tank wall temperature, external insulation temperature, temperatures at each end of the storage tank penetrations and supports, fill and drain line temperature, and fluid temperatures immediately ahead of each flowmeter will also be recorded. Flowmeters are located in the vent and outflow lines, outside the vacuum jacket, to measure flow rates and to aid in verifying a single phase flow. Pressurization flowrates will also be recorded.

F. FLIGHT TEST ARTICLE QUALIFICATION TESTING

This section defines the quality program to be used in the design, fabrication, and flight qualification of the flight test article and its backup hardward. The program covers all phases of contractor hardware performance including: development, procurement, fabrication, assembly, inspection, testing, acceptance, qualification, packaging, and shipment.

The application of these program control procedures assures the ability to perform timely and effective action to such conditions requiring action and to give evidence of conformance to the quality parameters as established for the program.

1. <u>Quality Program Management</u> - This task will define the lines of authority and responsibility in implementing the Quality Assurance Program.

a. *Quality Program Representative* - One individual shall be assigned responsibility for the Quality effort on the Orbital Program. This individual, Mr. James Tutchton, shall be assigned from the Advanced Program Quality Section of MMC which has direct access to the Director of Quality, Denver Division. He shall be responsible for performing or having performed the required Quality planning, inspection, laboratory analysis, calibration, qualification, acceptance testing, and delivery of the end item hardware. Control of the Quality Budget is included in this responsibility. Utilization of the expertise available from the Control Quality Organization in the performance of these tasks shall be under the control and direction of the Program Quality Representative. The Quality Representative shall be responsive to Mr. Paynter, the Program Manager, in all areas involving quality requirement of the hardware.

Figure II-8 depicts a typical hardware flow from procurement through delivery and outlines the areas of activity requiring Quality participation.

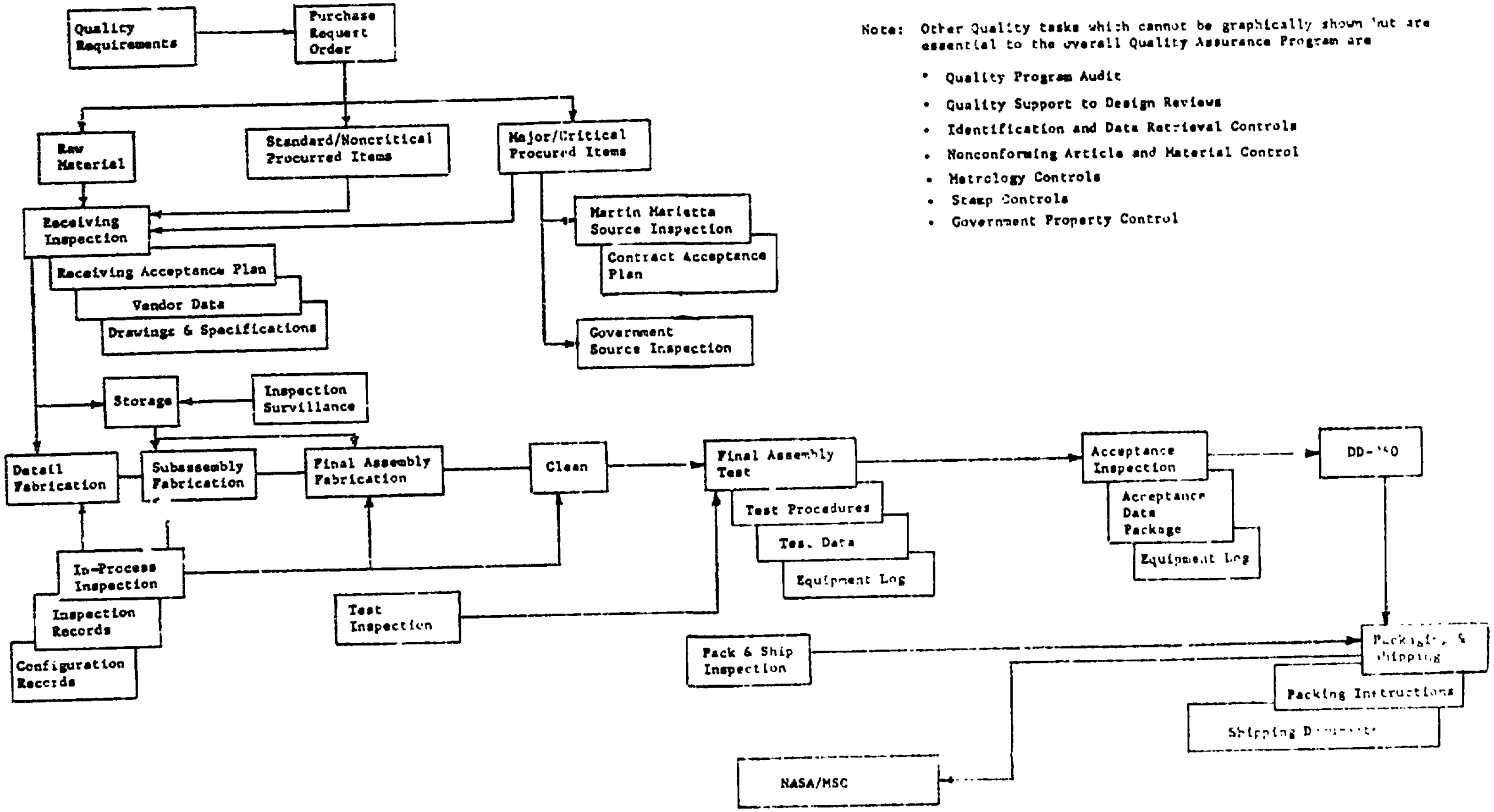
b. *Certification and Training* - Personnel responsible for process control, fabrication, and inspection will be trained and certified by Martin Marietta in accordance with the requirements of Quality Procedure 2.15 (Ref II-6). These personnel will be recertified, when applicable, at specified intervals or when performance dictates, to maintain skill levels to the required standards.

c. *Quality Status Report for Information* - Submitted on a monthly basis, these reports will be consolidated as a separate section in the Monthly Technical Progress Report and will contain:

1) Narrative comments regarding significant failures or problems encountered with procured and/or fabricated components;
2) Recommendations for corrective action and expected impact of the problems;
3) Status of such significant scheduled events as end item acceptance and major test completion.

2. <u>Design and Development Controls</u>

a. *Documentation* - Martin Marietta will maintain a documentation system for all aspects of the program that affect the quality of the product. Quality data considered essential to the program or developed by Martin Marietta as objective evidence of compliance to quality requirements will be made available to the NASA or AFQE



Note: Other Quality tasks which cannot be graphically shown but are essential to the overall Quality Assurance Program are

- Quality Program Audit
- Quality Support to Design Reviews
- Identification and Data Retrieval Controls
- Nonconforming Article and Material Control
- Metrology Controls
- Stamp Controls
- Government Property Control

Fig. II-8 Activity Areas for Quality Participation

Representative for review at their discretion. Martin Marietta standard procedures, the Quality Manual, and quality program directives will provide the direction and control of the Quality Program.

b. *Quality Support to Program Reviews* - Quality personnel will support program reviews, as required.

c. *Change Control* - The configuration management program will be conducted as required by Martin Marietta's Configuration Management Plan or Program Requirements.

3. <u>Item Identification and Retrieval</u>

a. *Identification* - Martin Marietta will utilize existing quality procedures that provide for and control the identification of piece parts, materials, and articles to which procurement, fabrication, inspection, test, and operating records are related.

b. *Retrieval* - The identification system will assure that applicable records relate to items specified in the identification list, so the items can be located and retrieved.

4. <u>Procurement Controls</u>

The Quality organization will have the primary responsibility for ensuring the quality of procured articles. Procurement quality activities will be performed to assure that: suitable suppliers are selected; the quality of incoming materials meets contractual, engineering, and program specifications; procurement documents contain quality and reliability requirements; materials are controlled during the manufacturing process by ensuring that they are properly identified and handled, and that adequate records of tests and inspections performed are available; and the supplier meets the requirements of the appropriate "Special Provisions of Purchase Agreement, Quality Assurance" clause.

5. <u>Fabrication Controls</u>

a. *Fabrication Inspection* - Inspection personnel will monitor fabrication operations, inspect all completed items, and perform in-process inspections as required to assure the proper level of workmanship and quality of the end item. The results of these inspections will be documented and the parts identified to provide objective evidence of Quality acceptance. Martin Marietta's

Quality Manual contain detailed descriptions of the fabrication and inspection cycle.

Discrepancies found during or prior to acceptance will be documented on a Quality worksheet, and the unacceptable items returned for corrective action. Items that cannot be reworked to drawings will be identified "withheld" and processed to the Material Review Board.

b. Structural/Mechanical Fabrication - All detail parts, including sheet metal, tubing, machined parts, plastics, and welded assemblies, will be fabricated in accordance with approved process plans.

6. Inspection and Test

Martin Marietta-fabricated articles will be controlled by a planned program of inspections and reviews conducted to ensure compliance with contract requirements during all phases of contract performance. These reviews will be conducted by Quality and will encompass all the technical documents used to fabricate and test the end items. In addition to establishing inspection requirements, these reviews provide assurance that the items can be built, inspected, tested, and that the engineering and contract requirements will be met.

a. Inspection Planning - All inspections of the end items, its components, or raw materials will be planned and documented to provide a complete record of the inspection, including by whom and the date it was performed. The planning will assure that inspections are performed in a logical sequence and at convenient points within the fabrication and test cycle, and will allow these operations to proceed consistent with good control practices.

The planning function will provide for coordination of Martin Marietta's scheduled inspections and tests with the designated Government Quality Representative.

b. Test Planning - All test operations will be performed in accordance with detailed step-by-step procedures as approved by Quality.

c. Assembly Inspection - Installations and assemblies will be accomplished in accordance with Quality-approved assembly plans. Prior to installing components, mounting holes and bracket

dimensions will be inspected for correct dimensions and locations per drawings. Quality will verify acceptable completion of component inspection and functional tests prior to installation of components and equipment into the circuit. All items in the process plan requiring Quality buyoff must be accomplished before proceeding to end item testing. To the extent practicable, each fabrication and assembly, inspection, and test operation will be traceable to the individual responsible for its accomplishment.

d. Test Inspection - Quality will control the validation of official test procedures, verify proper procedure configuration, and maintain status of all procedures completed, in work, or unaccomplished. Quality will review records, and witness acceptance tests. All variations, anomalies, or failures will be documented in the procedure history sheet section of the procedure.

e. Inspection and Test Records - Quality will maintain records of inspections and tests performed throughout the entire procurement, fabrication, and assembly process. The records will provide evidence that required inspections and tests have been performed on raw materials, procured parts, fabricated details, and completed articles. All quality data including vendor data, laboratory analyses, calibration records, nonconformance history, receiving inspection reports, and fabrication and assembly records, will be accumulated and maintained in retention.

7. Nonconforming Article and Material Control

All nonconforming material will be identified and a record completed and attached to the material. This form will describe the nonconformance, probable cause, and indicate the disposition prescribed by the responsible Quality Representative or by the Material Review Board. Discrepant items will be segregated and acted upon to ensure that the deliverable hardware contains only items meeting engineering and contractual requirements.

a. Identification and Routine Disposition - Hardware determined to be nonconforming will be identified and segregated. The initial identification is made by Quality personnel. A "Withheld" tag is attached to the discrepant item to denote pending disposition when required by procedure. When the disposition has been completely satisfied, including all retest and pertinent data review, a stamp indicating reacceptance is applied. Disposition of defective articles may be made without Material Review Board action. The dispositions that can be given are:

1) Complete to drawing - Items that are incomplete or can be restored to the original configuration will be corrected in accordance with drawings and specifications;

2) Recommend scrap - Items are obviously unfit for use or uneconomical to repair.

b. Material Review Board - Items that cannot be acted on routinely will be presented to a Material Review Board for disposition. The MRB shall consist of an authorized Quality Representative, an authorized representative of the Engineering organization, and a customer representative with acceptance authority.

Martin Marietta's representatives will be specifically authorized by Martin Marietta to act on material covered by this contract. Martin Marietta personnel will have been certified to participate in board activities. The MRB authority as related to suppliers is controlled in accordance with Section 10 of the Quality Manual.

The Martin Marietta MRB members will coordinate and recommend disposition prior to submittal to the customer representative. The MRB disposition requires the concurrence of all three members.

c. Material Review Areas - Nonconforming material will be removed from production flow and routed to a controlled area for processing.

d. Failure Reporting, Analysis, and Corrective Action - Martin Marietta will perform failure reporting, failure analysis, and corrective action using the procedures of Section 10 of the Quality Manual, which provide for early detection, reporting, and correction of conditions adverse to quality, and will provide control and corrective action of items that could degrade mission success.

8. <u>Packaging, Handling, Storage, Preservation, Marking, Labeling, Packing and Shipping</u>

Martin Marietta will assure control of packaging, handling, storage, and shipping functions in accordance with the contract specification and Quality Procedure 11.1. Shipping activities are monitored to ensure that items to be shipped will be properly preserved, packaged, and identified to prevent degradation during

transit. Documents and records accompanying each shipment will be verified to ensure conformance with established procedures and specifications.

9. Government Property Control

a. Detailed Quality Procedures - Detailed quality procedures for the control of government-furnished equipment (GFE) are contained in the Quality Manual, QP 5.3, Government-Furnished Property Receiving and Unpacking Inspection, and QP 5.5, Government-Furnished Property, Nonconforming, Control of, cover all aspects of Martin Marietta's responsibility in handling GFE. Martin Marietta will maintain records of GFE, including identification of the property, dates, types, and results of Martin Marietta inspections, and other significant events.

b. Contract Responsibility - Martin Marietta is responsible for receiving GFE and for reviewing data and documentation to determine that the hardware is acceptable for the intended use. Quality personnel will review inspection records and hardware to determine if shortages, damage, or unacceptable conditions exist. Functional testing will not be required. Martin Marietta will notify the customer representative of any GFE received that is unsuitable for its intended use.

III. PROGRAM PLAN: EXPERIMENT AS SECONDARY PAYLOAD

This plan describes a low-cost program to flight test the cryogenic test article as a secondary payload on the Titan IIIE/Centaur proof flight in January 1974. Although pursuit of this flight opportunity was terminated in January 1973, the program plan is presented here in detail since it represents a cost-effective program management approach that could be applied to other flight experiments. This approach reduces the extensive testing, test hardware, and paperwork usually required for flight hardware while taking advantage of available flight-qualified parts and subsystems.

The experiment baseline had been clearly established as a result of ten years of development effort including low-g experiment verification in drop tests and KC-135 aircraft. This experience allowed us to proceed directly from design and analysis to flight article fabrication with minimum technical and program risks, as dictated by the relatively short schedule length, to meet the planned January 1974 launch date.

The normal requirements for mockups, prototypes, engineering and development test articles were eliminated. Since the experiment was to be flown on a non-interference basis, flight-backup hardware was not required. Using this low-cost program approach, the cost for the entire flight test program was estimated at $1,600,000. As noted in Chapter II, the cost projected for the dedicated launch approach was about five times greater.

The experiment test module, as described here, consists of the dual-screen-liner (DSL) tank/feedline test article and required subsystems. The test module was to replace the Viking Lander Dynamic Simulator (VLDS) portion of the Viking Dynamic Simulator (VDS). The latter is the primary payload for the proof flight. The module was required, therefore, to simulate the mass properties and dynamic response of the VLDS so the controlled VDS test, as planned by NASA-LRC, would be unaffected by the replacement.

Because of the tight schedule, the test module could not be designed and fabricated in time for ground testing to verify its dynamic similarity to the VLDS. Rather, a Ground Vibrational Survey (GVS) model, similar to the orbital test module, was to be

built and delivered to General Dynamics in January 1973 for ground testing of the VDS. The ground vibrational tests were to be made with the VDS incorporating the VLDS and then the GVS model to provide a comparison of data. Similarity between the GVS and orbital test module was to be verified by analysis. The GVS model is described separately in Chapter IV.

A. TITAN IIIE/CENTAUR PROOF FLIGHT MISSION SUMMARY

The proof flight planned for January 1974 will be the maiden flight of the Titan IIIE/Centaur launch vehicle to be used for the Viking Program. A secondary objective of the flight will be to carry the VDS, a composite of the Viking Orbiter Dynamic Simulator (VODS), and the VLDS with an adapter trusswork. The VODS and the VLDS are simple, drum-like masses separated by the truss system. NASA-LRC plans to compare flight vibrational data, measured by strain gages attached to the legs of the truss network, to ground vibrational data to assure structural adequacy for the Viking Spacecraft.

In addition to the VDS, another smaller NASA-LeRC payload, the Space Plasma High Voltage Experiment (SPHINX), will be carried above the VDS and mechanically ejected to perform as a separate spacecraft during the Centaur transfer orbit coast.

As proposed in this chapter, the cryogenic DSL test module was to replace the VLDS, as shown in Figure III-1. The orbital test module, as shown, required its own truss system to support the SPHINX and required subsystems, communications, power, and thermal control.

Details of the Titan IIIE/Centaur vehicle and proof flight mission are summarized in Table III-1. As mentioned, the VDS, remains with the Centaur while the SPHINX is mechanically separated. As proposed here, the cryogenic test module would be part of the VDS; therefore, it would remain attached to the Centaur stage.

As presented in the table, the Centaur is placed into a near-synchronous Earth orbit at completion of the proof flight (about 8 hr after launch). It is at this point, i.e., after the proof flight objectives are satisfied, that the seven day orbital performance demonstration of the DSL tank and feedline system was to begin. Since the Centaur is a spent stage, the test module must have its own power supply and communications subsystems.

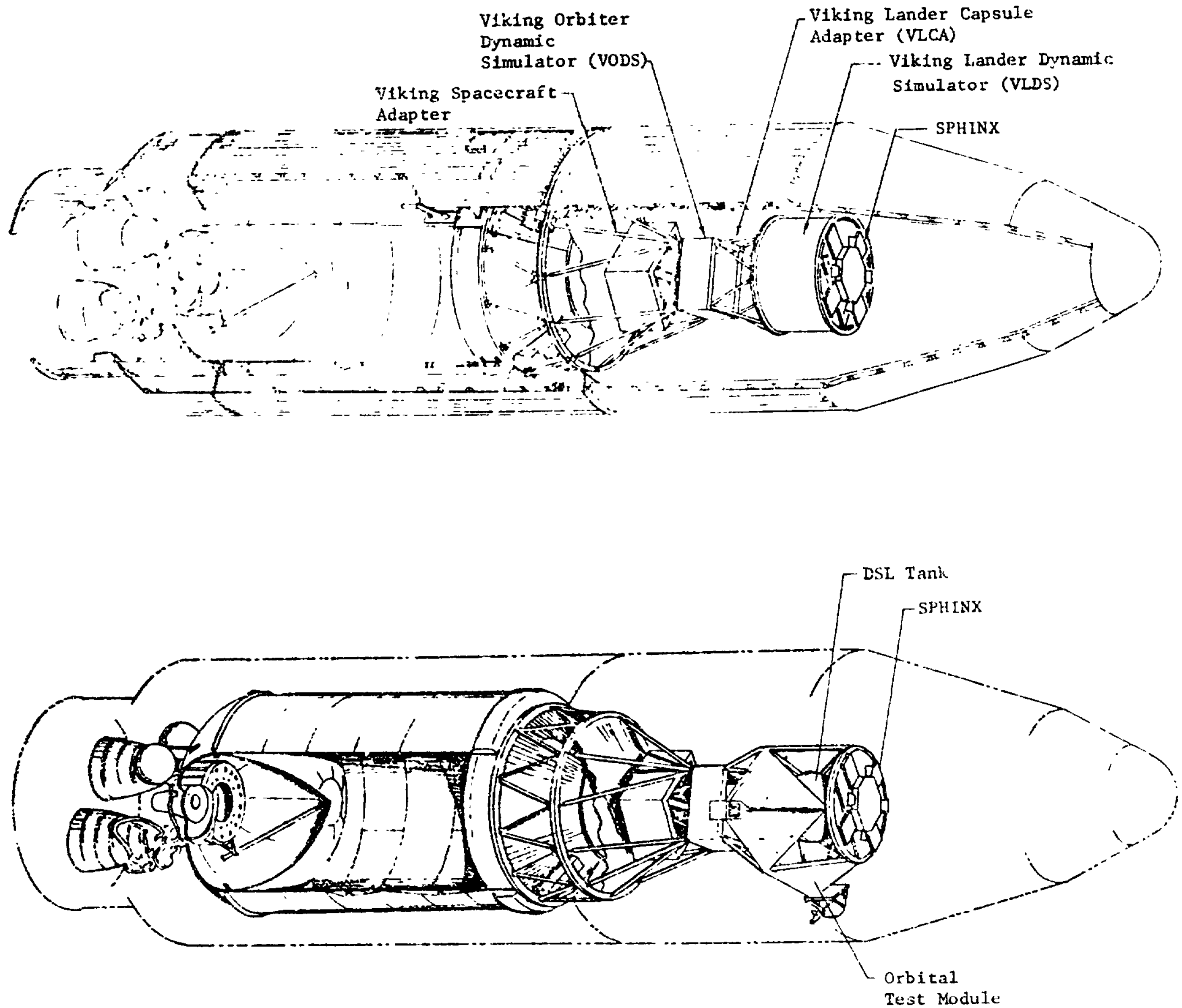


Fig. III-1 Proposed Replacement of VLDS by Orbital Test Module

Table III-1 Titan IIIE/Centaur Proof Flight Summary

CONFIGURATION	
Titan	IIIE
Centaur	D-1T-PF
Payload	- Nonseparable Viking Dynamic Simulator (VDS) - Separable - SPHINX Experiment - Nonseparable - Cryogenic Test Module (as proposed here)
LAUNCH PHASE	
Launch Date	January 6, 1974
Launch Mode	4 Centaur Burns, 3 Centaur Coast Periods
Launch Azimuth	105°
Launch Site	KSC Complex 41
CENTAUR TERMINAL ORBIT	
Perigee/Apogee Altitude	23,200-km/34,400-km (12,515/18,515-n-mi)
Orbit Inclination	30.4°
LAUNCH CONSTRAINTS	
SPHINX Experiment	Launch windows imposed, thermal and telemetry constraints
Orbital Cryogenic Experiments	None Imposed - Cryogenic experiment not activated until after Centaur proof flight mission objectives are satisfied (L + 8 hr)

1. Proof Flight Mission Objectives

The primary objective of the proof flight is to demonstrat the capability of the consistent Titan IIIE/Centaur system to support operational missions (Ref III-1). Secondary objectives are to demonstrate Centaur capability to perform both an operational two-burn mission with an extend d parking orbit coast and an operational three-burn synchronous orbit mission. A tertiary objective is to carry the VDS payload and inject the SPHINX spacecraft into a transfer orbit for a near-sychronous orbit. The final objective, as proposed in this chapter, is to deliver a cryogenic orbital test module into the terminal orbit. The cryogenic orbital test must be performed without compromising the proof flight mission objectives.

2. Launch Vehicle Characteristics

The Titan IIIE/Centaur vehicle consists of two five-segment solid rocket motors (Stage 0), the Titan first and second stage liquid propellant core sections (Stages I and II), and the Centaur third stage. The Centaur stage is an advanced high-energy model adapted for the Viking mission to be flown in 1975. Notable features include an updated electronics subsystem and a redesigned shroud assembly that encapsulates the entire Centaur and its payload.

Specific details for this launch vehicle are presented in References III-1 and III-2.

Titan IIIE/Centaur performance is shown in Figure III-2. To place a payload into the 23,200 km (12,517 n mi) by 34,400 km (18,515 n mi) elliptical orbit requires a characteristic velocity of 11,700 m/sec (38,300 ft/sec). Figure III-2 shows that the launch vehicle is capable of placing the 3568 kg (7850 lb_m) total payload into the elliptical near-synchronous orbit.

3. Proof Flight Profile

The proof flight launch occurs from KSC Pad 41 on an azimuth of 105°. The mission includes four Centaur burns to achieve four distinct orbits. The flight sequence and geometry of the orbits are shown in Figure III-3.

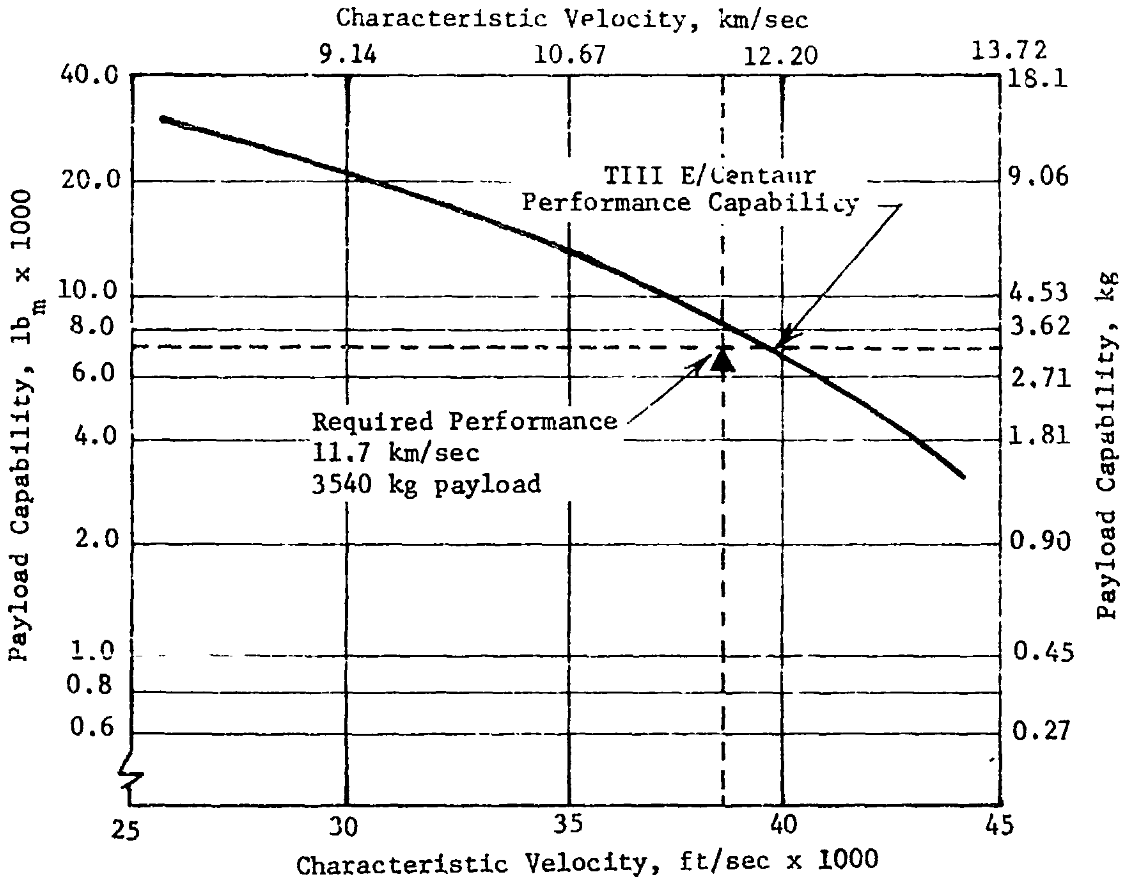
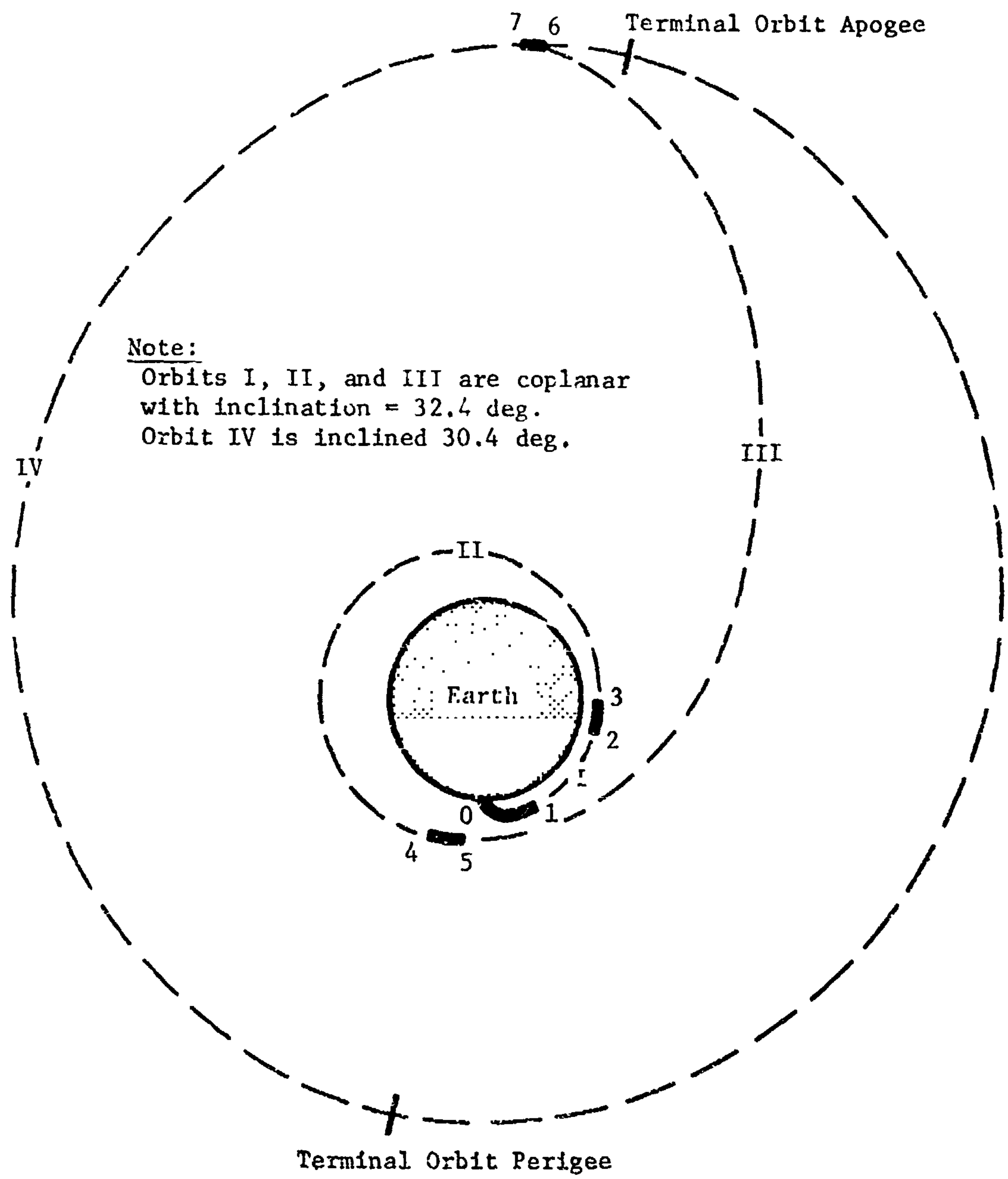


Fig. III-2 Titan IIIE/Centaur Performance



Flight Sequence	Orbit Sequence
0 LIFTOFF	I Parking orbit: 12 Min. Coast
1 MECO 1	II Intermediate Orbit: 80 Min. Coast
2 MES 2	III Transfer Orbit 5-1/4 Hr Coast
3 MECO 2	IV Terminal Orbit
4 MES 3	
5 MECO 3	
6 MES 4	
7 MECO 4	

Fig. III-3 Geometry of Proof Flight Orbits

The Centaur first burn (MES and MECO 1) of 136 sec duration injects the vehicle into a near 185 km (100 n mi) circular parking orbit inclined 32.4°. A 12-min coast (orbit sequence I) in the parking orbit is followed by a 50-sec Centaur burn (MES and MESCO 2) into an intermediate coast orbit inclined 32.4°. The 80-min coast (orbit sequence II) in the intermediate orbit is followed by the 183-sec Centaur burn (MES and MESCO 3) into a transfer orbit inclined 32.4°. The orbit has an apogee altitude slightly under synchronous orbit altitude. The transfer orbit (orbit sequence III) coast time is 5.25 hr. After 10 min of coast, the SPHINX is separated. The last Centaur burn (MES and MESCO 4) of 65 sec then places the Centaur in a 30.4° inclined terminal orbit approximating, but less than, a synchronous orbit size.

The transfer and terminal orbits and the plane change for the terminal orbit keep the SPHINX and Centaur from the undesired synchronous orbit space. The terminal orbit characteristics are, however, quite satisfactory for meeting the cryogenic orbital test module requirements since continuous ground contact is assured with either a 9.2 m (30 ft) or 26 m (85 ft) dia dish antenna of the Spaceflight Tracking and Data Network (STDN) system. This permits real-time telemetry transmission. The ground track shown in Figure III-4 is typical for 10 orbits (183 hr of flight) or approximately 7 days. Since the perigee and apogee altitudes almost cover an earth hemisphere, good station coverage is available for the telemetry reception.

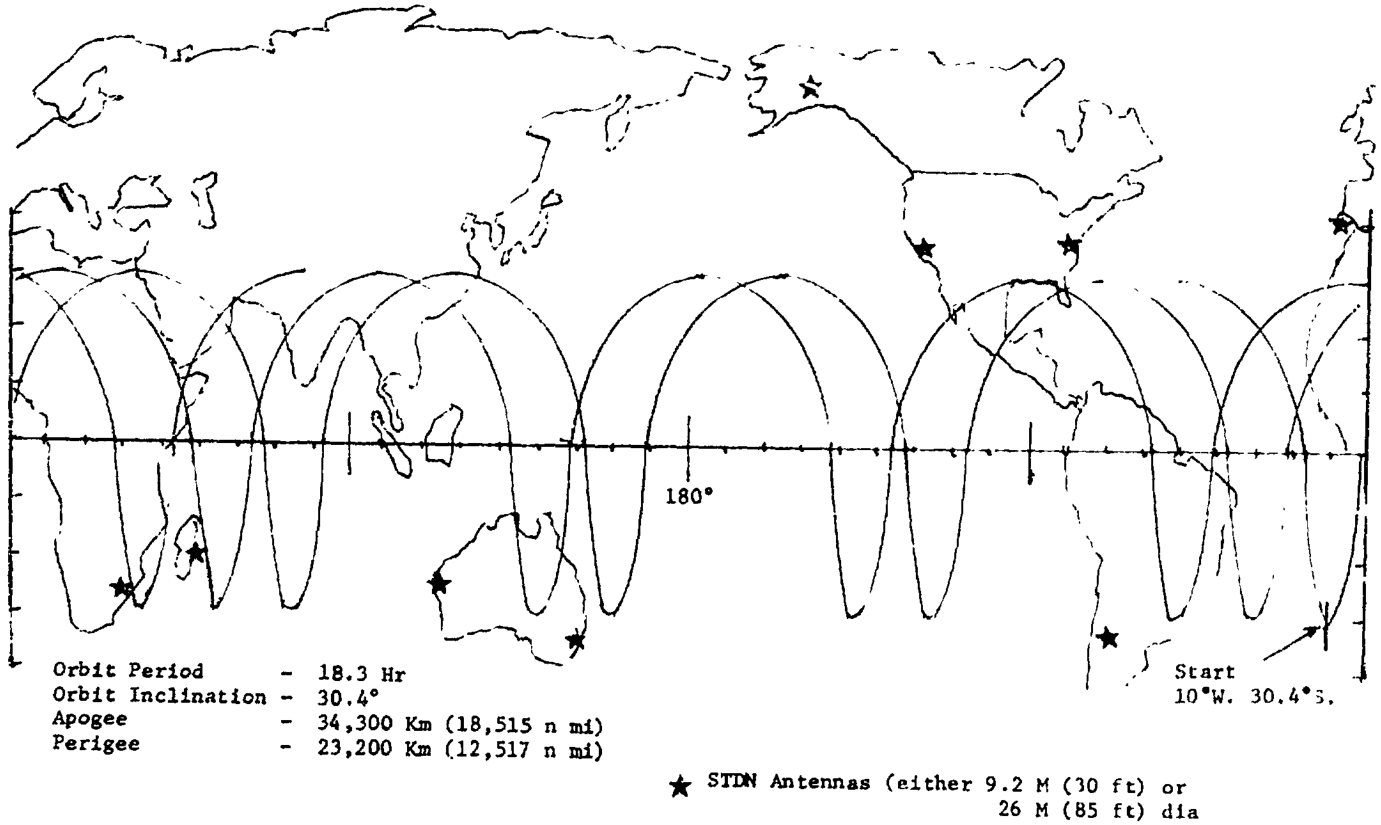


Fig. III-4 Orbital Test Module Ground Track - 10 Orbits (Typical)

B. EXPERIMENT DESIGN CRITERIA AND GUIDELINES

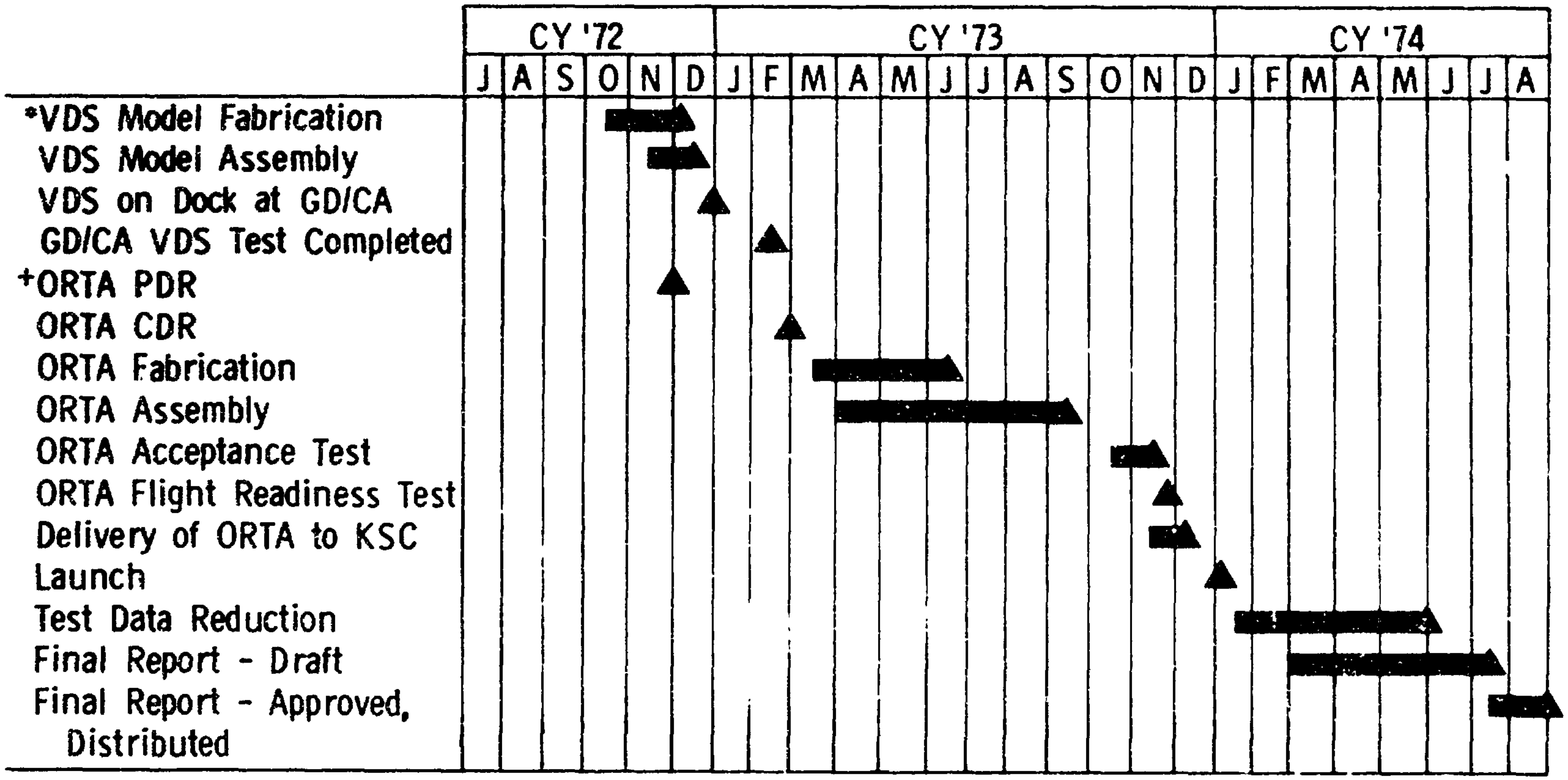
The test module was to demonstrate the DSL system's capability to provide: (1) liquid-free vapor venting for tank pressure control, (2) gas-free liquid expulsions, on demand, and (3) near-continuous control of the bulk propellant during the orbital experiment time period (7-14 days). The orbital test module includes: (1) the cryogenic test article (tank, feedline, and autogenous pressurization tank), and (2) support subsystems (power, communication, and thermal control). Recognizing the non-interference guidelines and other mission constraints, the orbital experiment was to be designed, fabricated, and qualified for flight within the relatively stringent schedule. Also, the experiment design was to be made using available components and qualified hardware, where possible, rather than identifying new and special equipment. We would design for low cost and to minimize testing and paperwork. The short schedule and the lack of mass and volume constraints permitted the low-cost program approach to be used.

The experiment criteria and guidelines, including design and interface requirements, are discussed in the following sections.

1. Schedule Constraints

The design, fabrication, qualification, and delivery of the orbital test module was to be accomplished in a period of about 16 months to meet the January, 1974 launch date. The program plan to meet this requirement is shown in Figure III-5. The initial milestone that had to be met was delivery of the GVS model to GDCA, San Diego, California, in January, 1973. The analysis, design, and fabrication of the GVS model, as a result, had to be concurrent with the design and analysis of the flight test article. Fortunately, the LO_2 tank/feedline assembly had already been designed under Tasks V and VI (see program schedule, Chapter I) to the level permitting Martin Marietta to meet this January date, with NASA-JSC concurrence. (The latter was granted by NASA-JSC on September 6, allowing four months to analyze, fabricate, test, and deliver the GVS model).

A second driver in formulating the program plan was the 5-month period to fabricate and assemble the cryogenic test article. The truss used in the test module would be identical to that for the GVS so it could be fabricated beginning in February. Since test article drawings could start being released in February also, tooling could begin. The detailed cryogenic assembly drawing release would be completed by March with the test module installation drawings released in May.



*Viking Dynamic Simulator
+Orbital Test Article

Fig. III-5 Phase C Program Schedule

Flight qualification of the test article had to be achieved with minimal testing to meet delivery to the Cape in early December, 1973. The approach was to design with large margins and safety factors, and use flight-qualified hardware. This was possible because of the more than adequate payload mass (3550 kg) goal and the low constraints on volume and packaging for the test module. The latter was to be qualified primarily by analysis with a minimum of testing. The test module would be acceptance tested at Martin Marietta before delivery to KSC. This was to be a functional test.

A preliminary design review (PDR) was held at Martin Marietta in December 1972. The results are discussed in subsection E.

Three critical design reviews (CDR) were scheduled for December, March, and April, 1973. The first was to review the flight truss; the second was to handle the flight test article; the third CDR was planned to review the orbital test module. In addition, a flight readiness review was planned for November or December, 1973 at the Cape.

The detailed work flow chart presented in Figure III-6 shows the parallel tasks and functions required to meet the program schedule. The key to satisfactory accomplishment of the tasks was use of the low profile management technique, whereby the MMC Program Manager and the JSC Technical Monitor would communicate frequently to affect changes and agree to modifications, without the need for any additional concurrence. Also, the MMC project team was to be comprised of personnel who would work more than one specialized task, i.e., the same people would be involved in the analysis, design, fabrication, and test.

2. Mission Constraints

The orbital experiment was to be conducted on a non-interference basis with the proof flight. The experiment could not impose constraints nor impair operations during pre-launch, launch, and during the 8 hr period following lift-off, during which the proof flight objectives were being met. The Centaur is a spent stage when the objectives are satisfied; its shroud has been removed and the SPHINX has been separated. There would be no attitude control capability and the Centaur would be in a weak tumbling mode. It would provide no power nor communications for the proposed orbital test module which would remain attached to the stage as part of the VDS payload.

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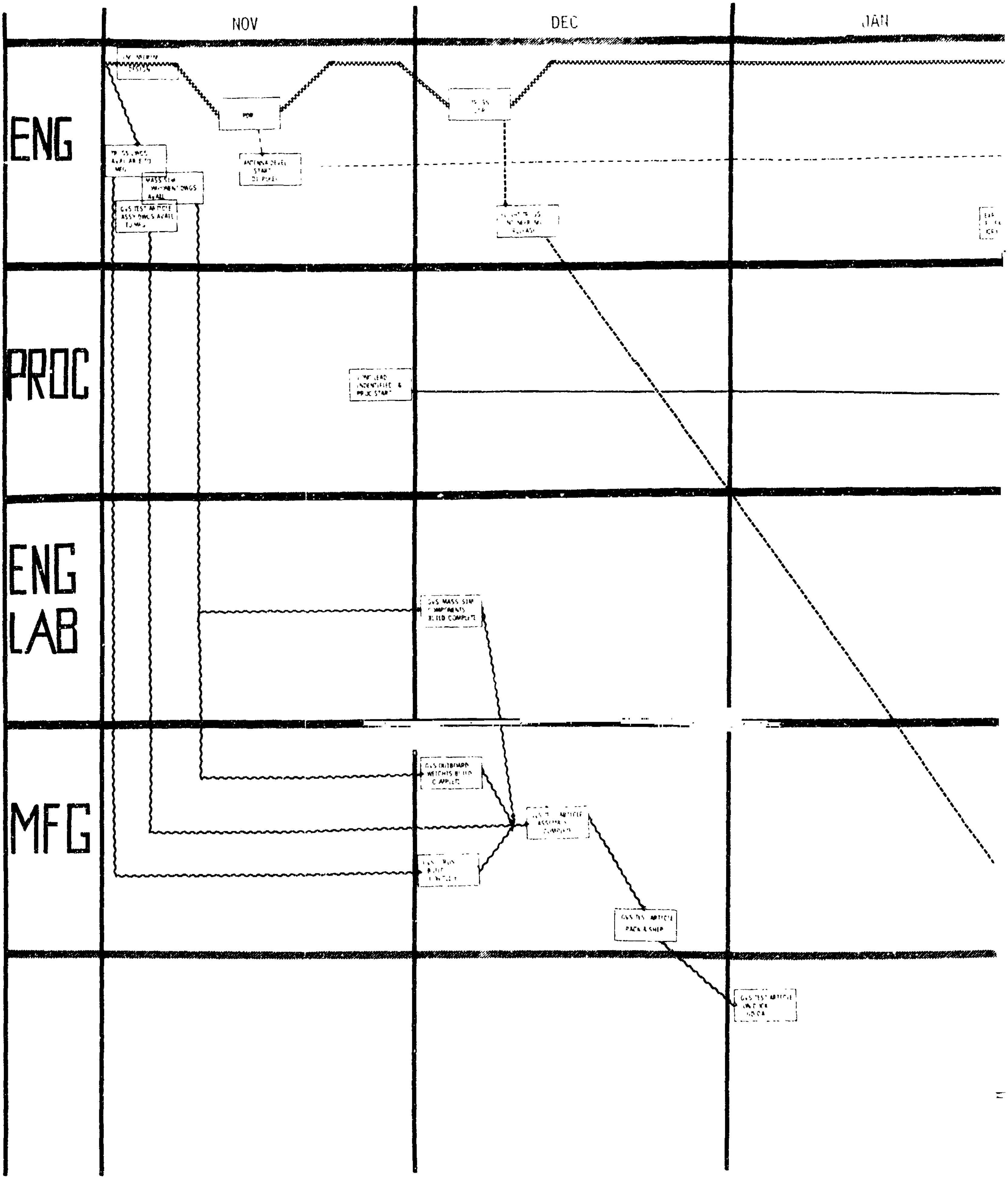

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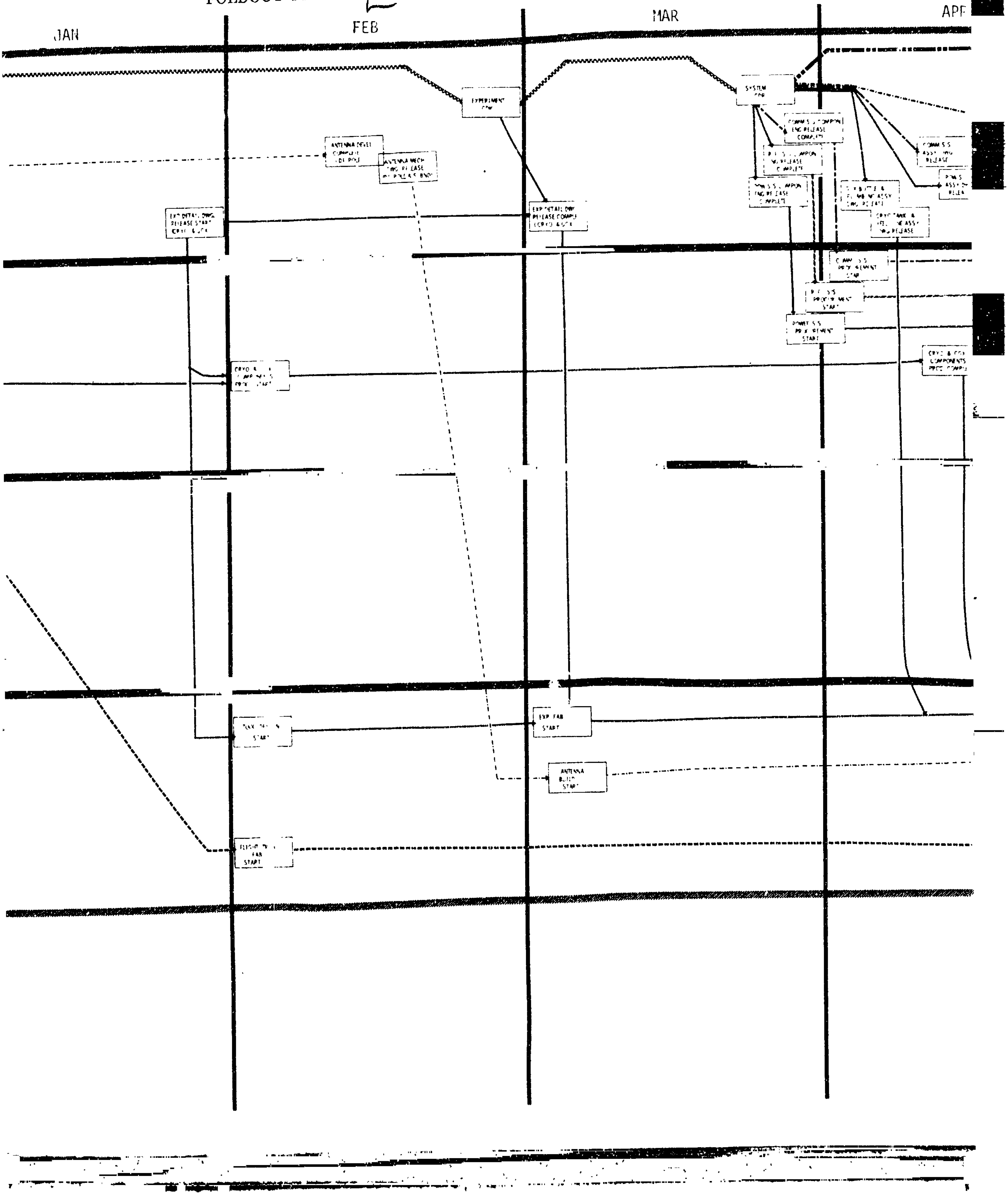

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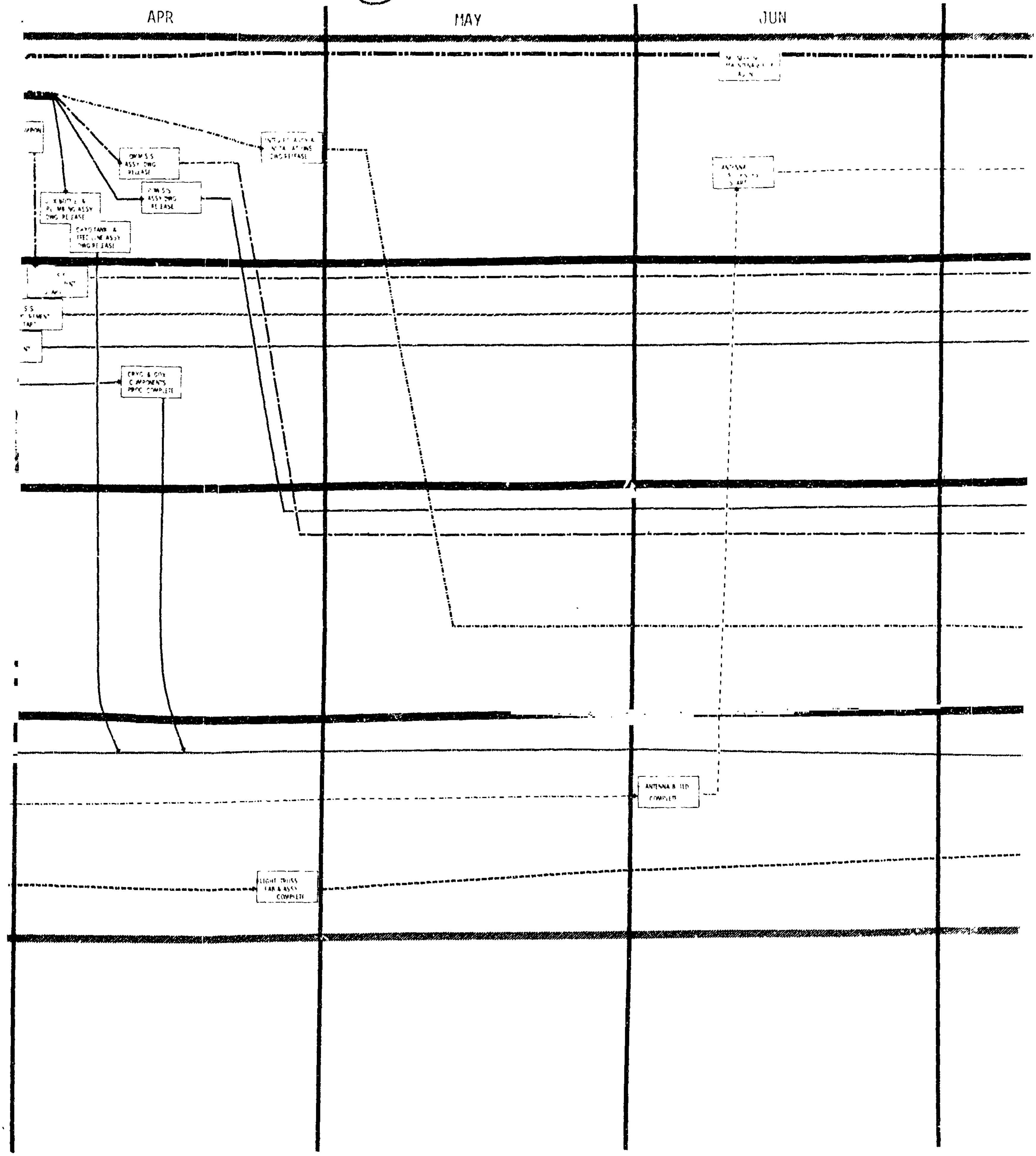

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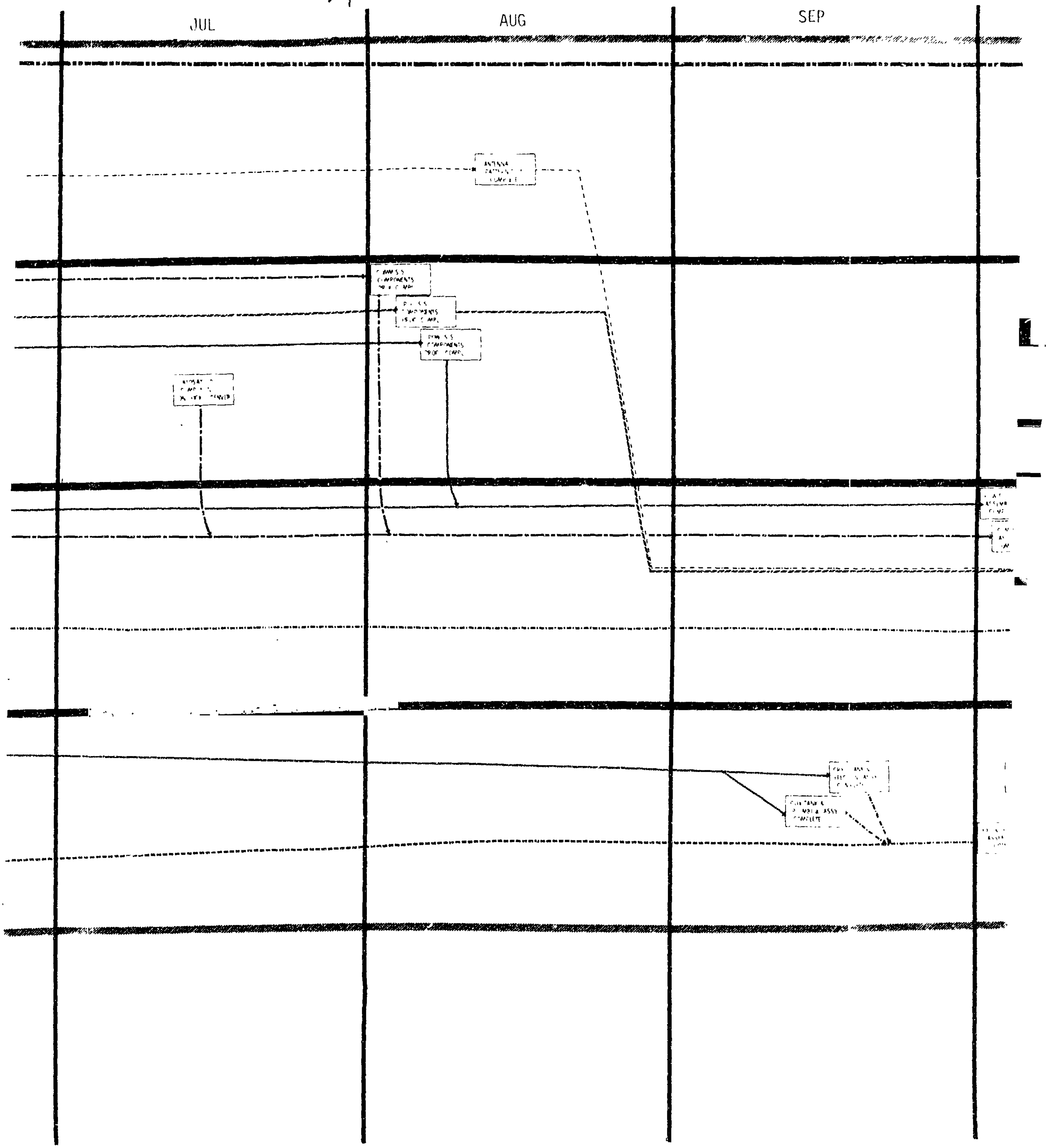

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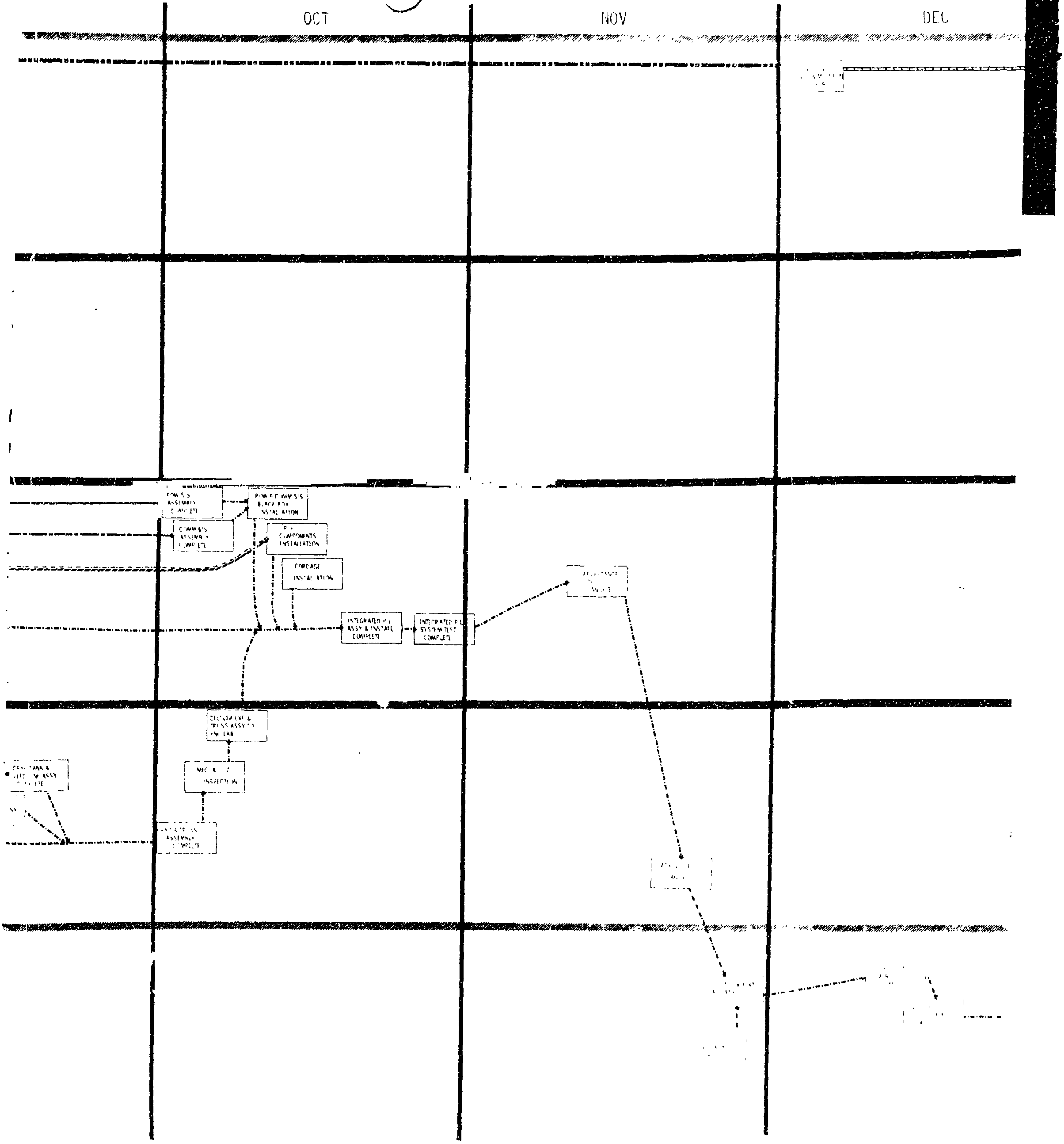

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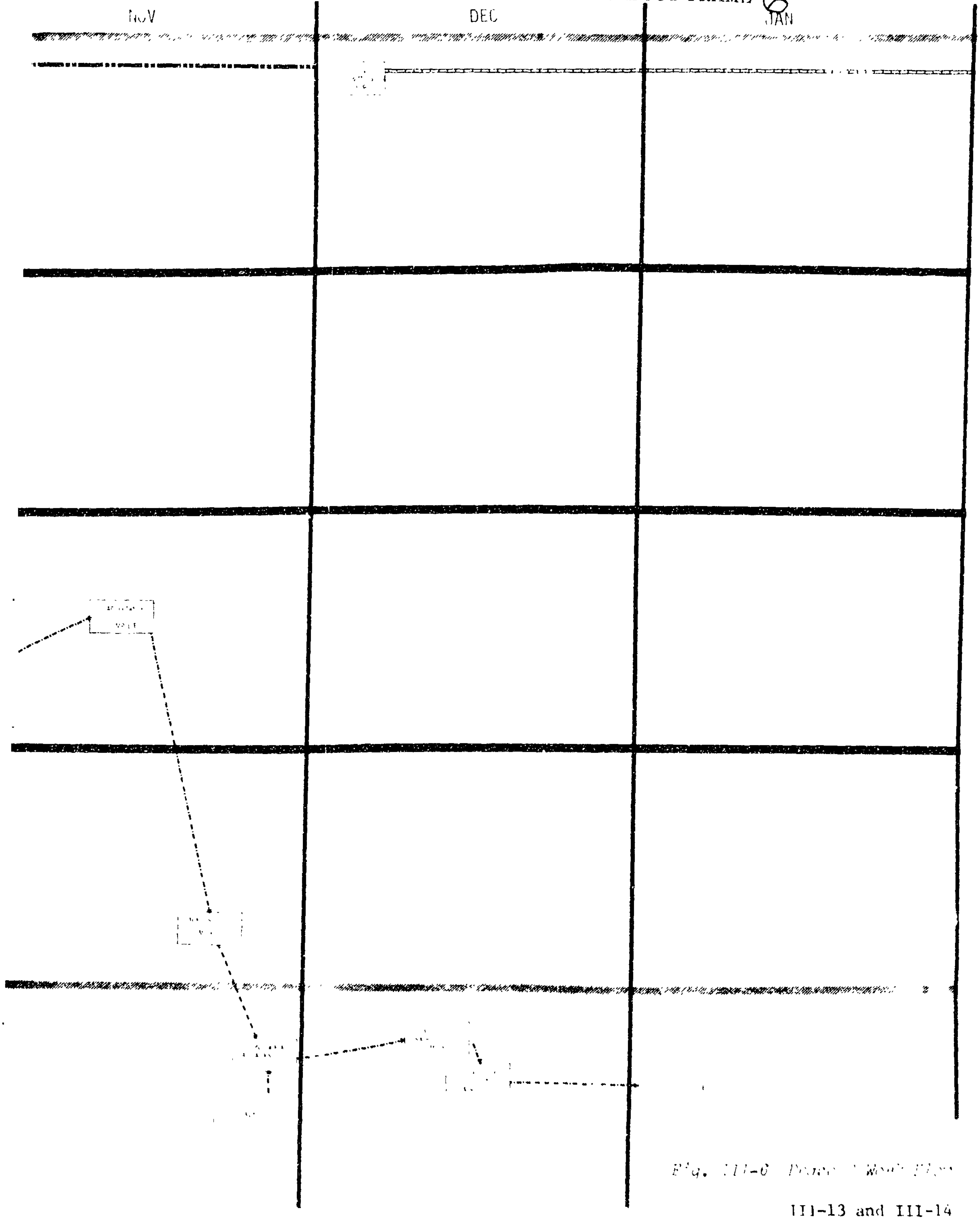

Fig. III-6 Phase [illegible] Work Flow

Activation and monitoring of the experiment would begin after completion of the Centaur objectives, or about 8 hrs after launch. The one exception mission would be the command receiver which would be activated before launch. The inclination and altitude of the proof flight terminal orbit are acceptable for communication contact and will provide an experiment duration of at least 7 days, as required.

As stated, the experiment would have to operate on its own internal power, as provided by batteries. It would also have to include its own antenna and communications subsystem for receiving and transmitting. Thermal control would be needed to provide the required environment for batteries, pressurization sphere, and the electronic packages.

The prelaunch non-interference dictated that the test module be loaded using its own portable dewar observing the pad accessibility constraints for the Centaur. Once loaded. the module would remain passive, except for venting and monitoring system pressure and liquid level, to vehicle launch. Venting would be into the Centaur shroud volume (to be purged with GN_2 on the pad). Access to the test module would be through an available door provided for servicing the SPHINX batteries (within 48 hr of launch). The module would remain passive during the first eight hours of the mission to prevent interference with the proof flight mission objectives. This meant no communication with the test module and no outflow or venting of fluid. In addition, to satisfy the NASA-LRC test objectives for the VDS, the test module was to contribute negligible slosh. The latter was to be assured by the loading and thermal insulation performance so that the LO_2 tank had less than a 5% ullage at launch. The tank was the only design worry since the feedline was dry at lift-off and the pressurization tank contained GO_2 only.

The test module would replace the VLDS dummy mass in the VDS payload. It had to simulate nearly identical mass properties and dynamic response of the hollow drum-like VLDS. These physical constraints are summarized in Table III-2. The first modal frequency of the test module was to be 40 Hz or higher. The load constraints imposed on the experiment during launch and flight are discussed in subsection D.

*Table III-2 Test Module Physical Constraints**

Weight	3550 kg		$(2555 \pm 100\ lb_m)$
C.G. Location:			
$C.G._x$	=	$1.01\ cm \pm 1.27\ cm$	$(0.4 \pm 0.5\ in.)$
$C.G._y$	=	$4.06 \pm 1.27\ cm$	$(-1.6 \pm 0.5\ in.)$
$C.G._z$	=	$572 \pm 1.27\ cm$	$(225.06 \pm 0.5\ in.)$
Inertia:			
I_x	=	$196 \pm 21\ kg\text{-}m^2$	$(467 \pm 50\ Slug\text{-}ft^2)$
I_y	=	$206 \pm 21\ kg\text{-}m^2$	$(490 \pm 50\ Slug\text{-}ft^2)$
I_z	=	$318 \pm 21\ kg\text{-}m^2$	$(760 \pm 70\ Slug\text{-}ft^2)$
Volume Envelope	=	342 cm (135 in.) dia cylinder, 158 cm (62.5 in.) high extending forward from Viking station 200 to station 262.5	

*VDS coordinates.

3. Interface Requirements

Interfaces included those with the VLCA, the SPHINX spacecraft, and the launch vehicle. The LO_2 tank was to be mounted in the center of an aluminum truss mounted on the VLCA. The truss provided a mounting for the SPHINX. The pressurant sphere, battery, and data acquisition and communication subsystems were mounted on the extremities of the truss system.

The test module attachment to the VLCA was at six places on the VDS at station 200.00. This attachment consisted of a shear pin and three 0.79 cm (5/16 in.) bolt holes at each location. Shear pin and bolt holes were to be located in the test module lower pads with conformance to the VLCA-to-VODS fit-check gauge.

The test module attachment to the SPHINX would be at three hard points on the forward triangular truss of the test module. Hole locations and mounting pad configurations are shown in Figure III-7. The truss system would support the 79 kg (175 lb_m) SPHINX

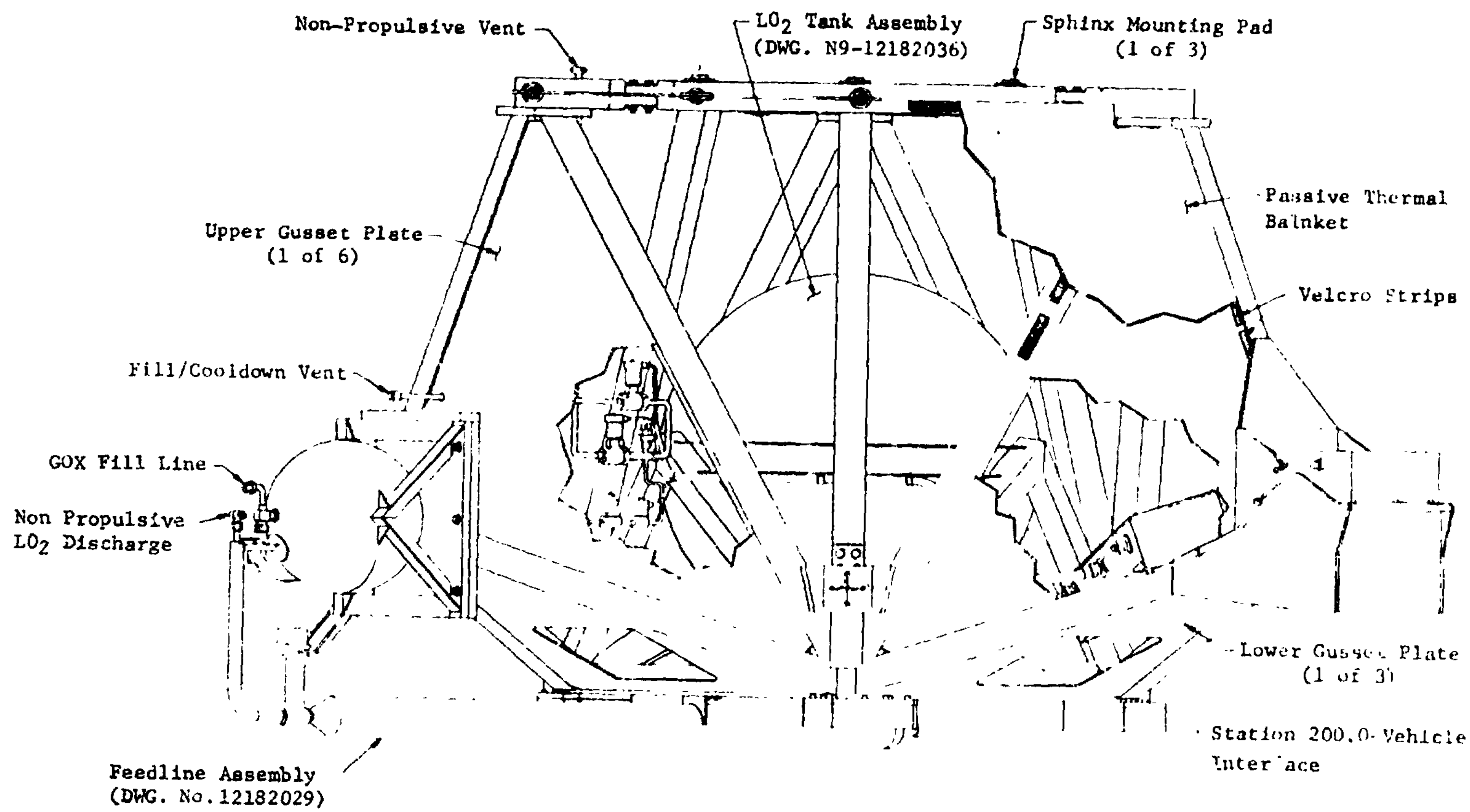

Fig. III-7 Flight Test Module Side View

throughout launch to separation. The latter is accomplished by pyrotechnic release of a preloaded spring. (The Centaur shroud has been separated previously). The SPHINX separation signal cable would be routed to the SPHINX along the test module truss from the Centaur. The separation signal is generated in the Centaur guidance system.

Two cables and four conductors would be routed from the Centaur to the data acquisition and communications bay of the test module to provide prelaunch monitoring of propellant level and propellant tank pressure. Routing lengths, locations, and connector configurations are shown in Figure III-7. After launch, no power or telemetry interface would exist between the test module and the Centaur or SPHINX systems.

The test module would require mechanical and electrical access after encapsulation in the Centaur shroud. Access would be provided through an existing 30.4 x 30.4 cm (12 x 12 in.) panel located between Centaur Standard Shroud (CSS) stations 2664 and 2649 with the centerline of the door on the + y axis of the CSS. This same panel would be used for module loading and monitoring of the test module prior to final countdown. At the completion of the loading sequence, mechanical and electrical systems would be disconnected manually inside the CSS and the fluid transfer lines would be capped.

A manually operated switch located in the transfer line disconnect area would be used to activate power to the command receiver prior to closing the panel cover. Thus, the experiment would be electrically active prior to launch, but in a power-down mode. After completion of the proof flight mission objectives, a ground station command would activate the experiment.

4. Design Criteria

The experiment test module was designed to meet the requirements for the low-cost management model using large margins for safety and structural integrity and using available, qualified components. It was a one-of-a-kind experiment where the same dedicated project team would be involved from design to flight test. The design criteria established for the orbital test module are presented in this section.

a. *General Design Requirements* - The orbital experiment module meets the following general design requirements:

1) A control system shall be established to ensure that the mass properties of the experiment module simulate the VLDS mass, c.g., and moment of inertia (see page III-16).

2) Vehicle axis is defined as follows:

    a) Yaw Axis - the vehicle Z axis;

    b) Pitch Axis - the vehicle Y axis;

    c) Roll Axis - the vehicle X axis.

3) Vehicle battery capacity shall be sufficient to satisfy power requirements for the 14 day time period.

4) All electrical/electronic equipment shall be designed to permit final checkout and verification with minimum removal from the modular assembly.

5) All equipment shall be mounted and packaged to allow reasonable access and convenient replacement.

6) The experiment module electrical and electronic equipment shall be installed and wired so that no unit would cause a malfunction of another due to conducted or radiated interference.

7) All components shall be designed to withstand an axial acceleration of 6.5g and a lateral acceleration of 2.5g.

8) All materials, valve seats, pressure transducers, temperature sensors, etc. shall be oxygen compatible.

9) All valves, meters, and instrumentation shall be flight qualified.

10) The storage tank shall be thermally protected to provide an environmental heating rate of approximately 1.58 W/m^2 (0.5 $Btu/hr\text{-}ft^2$).

11) Complex ground handling and ground fill procedures shall be avoided in every case.

b. Environmental Requirements

1) Prelaunch

   a) Temperature - 220°K (40°F) to 342°K (160°F) during transportation and storage. 270°K (28°F) to 310°K (100°F) during operations in uncontrolled areas at ETR. 295°K ± 3°K (72° ± 5°F) when in the Universal Environmental Shelter (UES).

   b) Contamination - TBD.

   c) Humidity - 5 to 95 ± 5% RH during uncontrolled operations. Less than 60% RF during operation in controlled areas.

   Handling Shock per MIL-STD 810 Method 516, Procedure V.

   d) Ambient Pressure - Corresponding to altitudes from sea level to 3,660 m (12,000 ft) for surface transportation and sea level to 10,680 m (35,000 ft) for air transportation.

   e) Vibration - Transient vibration to 5g peak during surface transportation and 10g peak during air transportation.

   f) Acceleration - Steady state acceleration to 3g during transportation.

   g) Salt Fog - An accumulation of up to 100μ per day on exposed surfaces near ETR.

   h) Precipitation - in the form of rain, snow, hail per NASA TMX-53872.

   i) Sand and dust per MIL STD 810B.

2) Launch and Ascent

   a) Thermal - TBD.

   b) Acoustics - Acoustic levels internal to the CSS not to exceed 145 db overall level with a spectrum shape per Table (TBD).

   c) Random Vibration - Random vibration transmitted from the booster vehicle not to exceed 4.0 g rms.

d) Pyrotechnic Shock - TBD.

e) Sustained Acceleration - Maximum sustained acceleration of TIIIE Stage I Cutoff of 4g.

f) Pressure - Atmospheric pressures from sea level to orbital attitude. To be represented by 10^{-4} mm Hg.

c. Structures and Mechanisms Design Requirements

Factors of safety, as defined below, would be used for design and test of the structure. In each case, the appropriate factor of safety would be applied to the critical (3 sigma) limit load. The "A" value of MIL-HDBK-5 shall be used as the allowable stress for the material.

	Factors of Safety	
	Yield	Ultimate
Module Truss Work	3	4
LO_2 Tank	3	4
GO_2 Sphere (Pressurization)	3	4

The communications and electrical compartment provide a mounting platform for the communications and electrical equipment. The communications compartment shall be instrumented with two temperature sensors to provide instrument beam temperatures. The communications and electrical compartment shall have mechanical attach points that interface with the support structure. The instrumentation and electrical compartment shall provide mounting surfaces for signal conditioners, timing correlator, current sensor, command receivers/decoders, relay assembly, power amplifier, commutator coder, and transmitter.

d. Communication Subsystem - The communication subsystem shall provide real-time measurement and transmission and timing correlation for the selected parameters. The system shall utilize a PCM/FM configuration as shown in Figure III-8. The telemetry transmitter shall provide for the transmission of the composite data with a minimum of 10 watts power output in the S-band range. The telemetry antenna would utilize the configuration shown in Figure III-8. The vehicle telemetry antenna pattern coverage over a sector area of a radiation sphere was TBD.

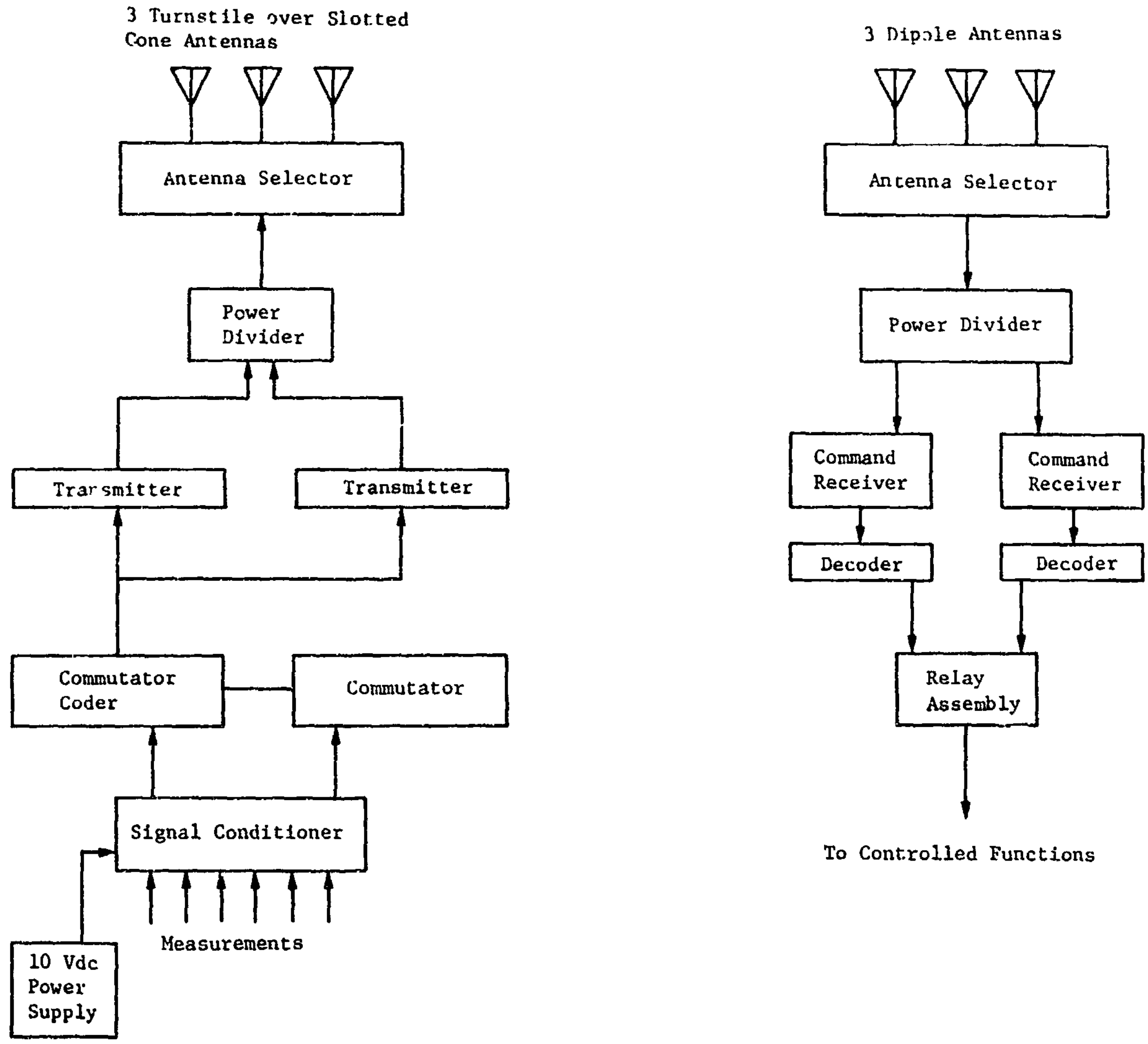

Fig. III-8 Communications Subsystem

e. Power Subsystem - The power subsystem would provide the following functions:

1) Electrical energy storage to power electronic equipment;

2) Switching of power equipment;

3) Electrical interconnections for all equipment;

4) Electrical power for thermal control;

5) Electrical power for the command receivers/decoders.

The main battery shall be instrumented with one thermistor to provide battery temperature.

The power subsystem include two batteries as the main power supply.

The main batteries shall provide nominal 28 VDC power. Table III-3 summarizes the load and margin for each battery. The power subsystem shall contain relay assemblies to provide for load switching.

Table III-3 Main Battery Load Summaries

Main Battery Equipment	Load (Amperes @ 28 VDC)
S-Band Transmitter Command System Instrumentation System Relays	TBD
Equipment Heaters*	TBD
Total	TBD
Battery Capacity Less 15 percent Margin	440 Ampere-Hours 66 Ampere-Hours
Battery Energy Available Required	374 Ampere-Hours TBD Ampere-Hours
Reserve	TBD Ampere-Hours (TBD percent)

*Estimated Heater duty cycles of TBD% on Equipment Heaters.

Power transfer to the command receiver shall occur at launch minus 48 hr.

Electrical interconnections shall be provided for all electrical and electronic components. Color coding, keying, and/or sizing shall be used to assure proper connector mating.

An isolated ground return system shall be used in the electrical subsystem, grounded at only one point, to obtain a single point ground system. Cable bundle separation shall be used to isolate high power, low power, RF, and pyro cables from each other. Separation, where possible, shall be at least 10.2 cm (4 in.) for cable runs of 0.61 m (2 ft) or more. Common connectors for pyro and other circuits are not acceptable.

Equipment voltage limit requirements are:

Main Power Subsystem	- Operating	- 28 ± 10% VDC.
	- *	- 32 VDC continuous.
	- *	- 36 VDC peak for 1 ms.

*Electrical/electronic equipment operating from these power sources shall be designed/procured to withstand the power levels with no damage or degradation.

f. *Attitude Control Subsystem* - The orbital experiment module shall be in an unstabilized mode during the operation of the testing sequence. The Centaur attitude control system shall operate through blowdown of the propellant tanks after the fourth Centaur burn. This procedure is required to maintain tumbling of the Centaur stage to a value less than two revolutions per minute about the Centaur pitch or yaw axis. Tumbling at a greater rate will result in loss of communication (ground-to-module).

g. *Thermal Control Subsystem* - The thermal control subsystem shall maintain battery and telemetry (TM) equipment temperatures within design limits throughout all phases of the mission. Thermal control shall be achieved primarily by passive means and be supplemented by active control where necessary. Passive thermal control involves the use of suitable combinations of insulation, thermal isolation mounting, thermal control coatings and finishes, thermal lagging, and conductive grease (to improve thermal contact between equipment and mounting structure). Active thermal control implies the use of thermostatically controlled heaters on selected equipment during the flight portions of the mission, and ground air conditioning of the entire vehicle during the prelaunch phases of the mission.

From a thermal control standpoint the test operations are divided into three phases as shown in Table III-4. Also shown are the principal problem areas associated with each phase and the general approach to their solution. The thermal control subsystem shall maintain module temperatures within the limits shown in Table III-5. A design margin of 269°K (25°F) below the upper limit, and 269°K (25°F) above the lower limit shall be used as a goal for passively controlled temperatures. The philosophy used in the preparation of Table III-5 was to maintain the listed components within the previously qualified temperature ranges. Thermal control shall be consistent with the internal heat dissipation rates shown on Table III-6 during the flight portions of the mission.

Table III-4 Thermal Sequence of Events

Flight Phase		Thermal Control Areas of Concern	Resolution
I.	Ground test operations from final assembly at Denver through calibration at KSC	Possible RF system overheating due to internal heat dissipation	Provide forced-convective cooling with ground air conditioning (in buttoned-up configuration)
II.	Prelaunch checkout, L - 48 hr (command receiver) through Launch	Equipment overheating due to internal power dissipation and/or ambient heating	Use thermal mass of equipment compartment and structure to partially absorb internal heat
III.	Ascent and orbit	Maintaining equipment temperatures within design limits. Environmental uncertainties	Provide passive and active thermal control as required. Define thermal-environmental models to bracket the manifold of possible environmental conditions during the module flight

Table III-5 Design Temperature Limits

Electronics Compartment	256 to 344°K (0 to 160°F)
Primary Battery Compartment	278 to 305°K (40 to 90°F)
Antennas	
Command	156 to 450°K (-180 to 350°F)
TM (S-Band)	156 to 450°K (-180 to 350°F)
GOX Pressurization Sphere	TLD
Cabling	
Coaxial	218 to 353°K (-67 to 175°F)
Stranded	208 to 353°K (-85 to 175°F)

Table III-6 Equipment Heat Dissipation

Electronics Compartment	
Standby	86.0 watts
Transmitting	*176.0 watts
Primary Battery Compartment	
Standby	** 12.9 watts
Transmitting	** 26.4 watts

*Standby and transmitting.
**Assumes 15% of load in heat dissipation.

Forced-convective cooling of the RF system would be provided during prolonged ground testing with power on. Ground air conditioning shall be provided to prevent equipment overheating, flight battery degradation, and thermal preconditioning (prior to launch).

h. *Electrical/Electronic Test Support Equipment (TSE)* - The Electrical/Electronic TSE for the orbital experiment module must support testing at the subsystem, system, and selected component level at multiple test areas at Denver and at the launch site. These require that the equipment be adaptable and mobile. The TSE will provide the capability for malfunction isolation to the replaceable black box. The TSE shall be capable of supporting the following levels of tests:

1) Individual major module assemblies and their subsystems;

2) Incremental tests during the buildup of major assemblies into a complete module;

3) Integrated tests of the module;

4) Integrated tests of the module after mating with VLDS adapter and mounting of the SPHINX equipment.

The TSE shall be capable of supporting test activities at Denver in Vehicle Test Area AMT Building and at KSC in the Assembly and Checkout Area SAEF Building.

The TSE shall be located in a van or trailer so it can be easily transported between test locations. The van include heating and air conditioning equipment, AC/DC power distribution equipment, lighting, voice communications equipment, shelving, cabinetry, and work benches. Certain electronic equipment shall be mounted in electronic enclosures in the van. Other equipment will be stored in cabinetry and on shelving in the van to be used as required for checkout support at the test areas. Provisions shall be included in the van for tiedown of these equipment during transit.

The TSE shall include the following major elements:

1) Rack Mounted Equipment - This equipment shall include the permanently mounted MMC fabricated and commercial panels, chassis, and associated cabling necessary for functional checkout of module subsystems. There will be approximately seven racks to house equipment which was not required for testing external to the van. Typical rack mounted equipment will be a pyrotechnic panel, battery checkout panel, checkout and monitor panel, receivers, electronic counter, tape recorder, strip-chart recorders, signal generator, discriminators, calibrators, power supplies, and communications equipment.

2) Checkout Support Kit Assembly - This is an assembly of standard test equipment, meters, simulators, calibrators, checkers, and adapters, required to complete equipment needed for vehicle checkout, calibration, and malfunction isolation. Most of this equipment shall be used on benches within the van and outside in the vicinity of the experiment module. However, if deemed practical, some test equipment could be rack-mounted within the van.

3) Interconnection Panels - The van shall incorporate electrical and RF interconnection panels to permit the connection of van external cables.

4) Antennae and Transmission Lines - This equipment shall be used outside the van for RF transmission between module/RF systems and the RF checkout equipment within the van.

5) Interconnection Set, Ground Equipment - Electrical - This set shall contain the various cable assemblies required to interconnect the TSE van with the module. Cables shall also be provided to interconnect equipment which is not van-mounted to the van or the vehicle including facility and portable power sets.

6) Cable Set, Van Internal, Power AC/DC - This cable set shall be required to distribute AC and DC power feeders from points of distribution to equipment items and work locations within the TSE van.

7) Van Interconnection Cable Set - This cable set shall provide cables for interconnecting TSE within the van and for connecting van TSE with the van wall-mounted panel assemblies.

The TSE shall incorporate the following major functional checkout capabilities:

1) Instrumentation - The equipment for instrumentation checkout shall provide the capability to simulate transducers and sensors and stimulat· various components for checkout. The TSE for instrumentation shall consist of the above simulators, oscillators, strip chart recorders, discriminators, calibrators, a tape recorder, patch panels, a decommutator, cables, counter, digital volt meter (DVM), and an X-Y plotter.

2) RF Equipment

   a) Telemetry Transmitter - The checkout of the telemetry transmitter shall require provisions to measure its RF power output, output frequency, and carrier deviation. TSE to conduct this checkout shall consist of a receiver, watt-meter or power meter, test oscillator, counter, termination, coupler, cables, checkout antenna, and a spectrum analyzer.

b) <u>Antenna</u> - TSE shall be provided to check antenna system cable attenuation, RF component, antenna and cable VSWR, and component insertion loss. Equipment to accomplish this shall include a sweep generator and plug-ins, a network analyzer, detectors, couplers, terminations, and cables.

c) <u>Command</u> - The TSE shall include provisions to transmit modulated commands via closed loop RF transmission to the module command system. The TSE shall similarly provide the capability for verification of command receiver sensitivity, receive frequency bandwidth and decoder outputs.

d) <u>Battery</u> - The TSE shall be capable of mating with the module umbilicals as well as battery connectors to checkout and simulate the airborne battery system.

i. Assembly, Handling, and Support Equipment (AHSE) - AHSE consists of equipment with the capability to lift, transport, adapt for weighing, balancing and c.g. determination, support, store, and align the module and components. AHSE shall include the following:

1) Assembly fixtures for use during module buildup and equipment installation;

2) Handling fixtures to lift, move, and rotate module components;

3) Transportation equipment for relocation of the module and components between assembly, test and operations facilities;

4) Protective containers for shipment of the module and detachable assemblies or components;

5) Special test fixture adapters to provide the module positioning and attachment to standard test equipment;

6) Access ladders and platforms for use during module assembly, test and servicing.

5. <u>Program Control Requirements</u>

The program management and control system proposed for the orbital test program were based on Martin Marietta's experience from integrating scientific payloads, transtage, Skylab, Viking and

building specialized one-of-a-kind hardware. The approach was similar to that used in recent Martin Marietta programs, including payload integration for the Titan III, C-5 and C-14 programs, Skylab, and the X-24B modification program. The cryogenic orbital experiment program was to be patterned more nearly to the X-24B program with regard to program control, where management would be a joint effort between MMC and NASA program managers. Timely and accurate communication between the two was a necessity to meet the program objectives in a cost-effective manner.

Martin Marietta would provide the necessary resources to design, develop, build, and deliver the integrated flight payload, consisting of the cryogen acquisition/expulsion system and the required support flight subsystems. Martin Marietta would also design and build additional hardware required to perform development/qualification testing as well as that required to perform ground testing, checkout, handling, transporting, and pre-flight operations. On-site support services would be provided during the launch, orbital operation, and data evaluation phases.

a. *Management Control* - Management control would be cost-effective as dictated by the low cost, one-of-a-kind nature of this program. The program manager would apply those controls that were economical while providing the visibility and close-coupled MMC/NASA response required. Our experience on this type of program had proven that these efficiencies could best be achieved by primary dependence on manual systems accomplished on-program by team personnel, as directed by the program manager.

The program would function as a separate team supported by the central engineering organization. All work effort and associated budgets required to accomplish the tasks defined in the Statement of Work (SOW) would be formally issued by the program manager to the program team members by an Operations Directive (OD). Each task leader would be responsible for controlling budget and schedule performance within the areas defined by the specific task OD. The functional task leaders would control their work on a day-to-day basis with the project engineer. All detail schedules would be statused weekly for the weekly telecon review with the JSC contract monitor so that any problems, technical, management and cost, would be identified for joint resolution by the program manager and contract monitor.

b. *Schedule Control* - The status and review functions would be performed daily by the project engineer and reviewed weekly, at least, with the program manager. Through daily observation and discussion, the NASA and Martin Marietta representatives would

quickly proceed toward completion of task requirements. Requirements would be assessed in real-time with particular attention given to sequence and the interfacing of milestones. Potential schedule problems would be dealt with as identified and corrected in the agreed manner.

c. *Cost Control* - Detailed in-house budgets would be established by the cost management representative with each functional department program lead and assigned by OD after approval by the program manager. Budgets would be established separately for manhours, labor dollars, travel, material and computer charges, and time-phased by month. The lead personnel would control his task manpower by providing personnel with prepunched time cards. This would assure the proper charging of labor to the program and allow him to add or remove personnel quickly for efficient program performance.

The cost management representative would utilize daily labor reports that identify hours by employee name and total dollars charged to each of the tasks to compare the planned expenditures with actuals, identify trends, variances, and discrepancies. Cost performance would be compared to schedule progress to identify problems.

d. *Configuration Management* - A configuration management plan would be prepared to establish a system for controlling the release of engineering drawings and changes. The system would also provide hardware status accountability and depict the actual hardware configuration at any time along with the incorporation point of any changes. A change control system would assure a complete and accurate impact assessment of any proposed changes. All changes to released engineering, contractual, or other controlled documents would be processed through this change control system prior to incorporation.

Changes identified by fabrication, assembly, design, and test personnel would be acted on by the program manager. He would have sole authority to approve changes for the contractor and would coordinate those changes requiring JSC approval with the NASA program manager.

The configuration of the orbital experiment hardware at delivery to KSC would be identified in report form and by the release records and the quality control logs. The logs would be verified by the reliability and quality assurance representative as

containing the current changes identified by the release records. An acceptance package would be prepared and presented for NASA approval based on this information. This single configuration record controlled by the program manager would provide for the desired control and identification of the hardware configuration at any given time during the program.

6. Analysis and Documentation

In the interest of economy, it was desirable to provide a design that required minimal testing. Therefore, analyses were to be conducted to substantiate the existence of design and performance margins suitable to justify the exclusion of testing. These analyses would be submitted at the first Critical Design Review (CDR) in lieu of tests. Such analyses would be documented and submitted in accordance with the DRL line item.

Failure Modes and Effects Analyses (FMEA) would be performed for critical hardware and submitted to NASA in accordance with the governing Data Requirements List (DRL) line item. Preliminary system level FMEA results would be presented at the CDR. For critical electronic systems, worst case analyses would be performed to demonstrate that possible parametric variations would not degrade system operation such that mission objectives were not met. Other analyses such as power consumption, RF link, stress and environmental control, would be performed as a normal part of the design process to reach the preferred configuration. The results of such analyses would be made available to NASA upon request and reviewed at PDR, CDR, Readiness Reviews, but would not be DRL line items.

Martin Marietta would furnish all data items identified and described in the DRL, NASA Form 1106. The data items would be prepared in accordance with the Data Requirements Descripition (DRD), NASA Form 9. Where practical, internal documents would be utilized to meet and/or supplement the requirements specified in the applicable DRD. In addition to the data identified and described in the DRL, other supplemental data would be furnished as required by the SOW and attachments.

7. Quality Assurance

The Quality Program Plan presented describes the Martin Marietta Aerospace Quality Assurance program to be used in the orbital test module design and fabrication.

The following documents for the date and issue shown, shall form a part of this plan:

MIL-I-45208A	Inspection System Requirements 16 December 1963
MIL-C-45662A	Calibration System Requirements 9 February 1962
USAF Specification Bulletin NR-515	Control of Non-Conforming Supplies.

a. *Program Management* - The Quality Department shall implement Martin Marietta practices and procedures in their existing form or shall adjust them to provide the inspection system necessary to meet program requirements, in accord with the program manager. The system planned for this program shall comply with the requirements of MIL-I-45208A.

Martin Marietta shall provide a system for planning, managing, implementing, and assessing the quality program to assure that quality requirements are identified and satisfied through all phases of contract performance. The Quality disciplines that shall be addressed are: project management and administration; quality engineering and planning; supplier selection, performance and evaluation; inspection and acceptance; configuration assurance; test assurance; non-conforming material control and corrective action; quality data and records collection; and skill certification.

b. *Organization* - The Research and Development Laboratory (RDL) Quality Program Manager shall be responsible for all Quality activities associated with the orbital test program. The Quality Manager shall be the point of contact for Quality activities. The R&D Quality Section of Quality Engineering shall perform the quality tasks in support of the effort associated with the program.

c. *Initial Quality Planning* - RDL Quality shall conduct complete review of the requirements of this program and shall identify and make provision for special controls, processes, test equipment, fixtures, tooling, and skills required. This effort shall be accomplished by the issuance of project directives to the affected Quality line organizations to identify specific quality requirements to be accomplished.

d. Work Instructions and Inspection Record Forms - The Quality organization shall assure that instructions for all work affecting quality, relative to purchasing, material handling, machining, assembling, fabrication, test, installation and modification are provided.

For all new or refurbishment fabrication of details, subassemblies, and assemblies fabricated, an Inspection Record Form shall be utilized to assure conformance with engineering drawings and quality requirements and provide objective evidence of hardware status and Quality Acceptance. End-item level test procedures, generated by Engineering shall be reviewed by Quality for compatibility with the drawing and specification requirements and for identification of specific quality inspection points.

e. Inspection and Test Records - Quality records, such as supplier data, receiving inspection records, laboratory analysis, calibration records, in-process fabrication and assembly records, rework and modifications, in-process data, and final end-item test records, shall be maintained for this program as objective evidence of hardware Quality acceptance. These records shall indicate the nature and number of observations made, the type and number of deficiencies found, the quantity approved or rejected, and the nature of corrective action taken, as appropriate. Items shall be traceable to the individual who accepted the operation or article.

f. Corrective Action - The Quality organization shall assure that prompt action is taken to correct conditions that could result in the submittal to the government of supplies and services which do not conform to: (1) the quality assurance provisions of the item specification; (2) inspections and tests required by the contract; or (3) other inspections and tests required to substantiate product conformance.

The Minor Discrepancy Report form will be used to document minor variations of the hardware from the drawing in those instances that the "as-built" drawings do not reflect the true condition of the hardware.

8. <u>Facilities and Standards</u>

1) <u>Drawings, Documentation and Change</u> - The Quality system shall provide procedures to assure that the applicable drawings, specifications, and instructions required by the contract and authorized changes thereto, are used for fabrication, inspection and testing, and quality acceptance.

2) Measuring and Testing Equipment - The contractor shall provide and maintain gages and other measuring and testing devices necessary to assure that supplies conform to the technical requirements. To assure continued accuracy, the devices shall be calibrated at established intervals against certified standards which have known valid relationships to national standards. If production tooling, such as jigs, fixtures, templates, and patterns is used as a media of inspection, such devices shall also be proved for accuracy at established intervals. Calibration of inspection equipment shall be in accordance with MIL-C-45662. When required, the contractor's measuring and testing equipment shall be made available for use by the government representative to determine conformance of product with contract requirements. In addition, if conditions warrant, contractor's personnel shall be made available for operation of such devices and for verification of the accuracy and condition of the measuring and testing equipment.

3) Procurement Controls

a) Supplier Controls - Controls shall be established for suppliers of hardware and material to assure that technical and quality requirements are identified and are maintained from start of the procurement cycle through delivery to Martin Marietta. Systems presently in effect shall be used for qualified supplier selections, review by Quality personnel of purchase orders to assure inclusion of quality requirements, source inspection, as nec ssary, and verification of supplier ratings.

All purchase requisitions for material, components, subassemblies purchased or subcontracted by the Engineering Electronics Laboratory, shall be reviewed by Quality Project for coding of quality requirements.

Hardware and materials procured by Martin Marietta Corporation shall be subjected to receiving inspection prior to release for fabrication or installation. This inspection shall include identification, visual inspection for damage, functional testing as applicable, and compliance to purchase order requirements.

b) Purchasing Data - All documents and referenced data for purchases shall be available for review by a government representative to determine compliance with the requirements for control of such purchases. Copies of purchasing documents required for government inspection purposes shall be furnished in accordance with the instructions of the government representative.

Manufacturing Control

a) Materials and Materials Control - Subcontracted or purchased supplies shall be subject to inspection at source and/or after receipt, as necessary, to assure conformance to contract requirements. This inspection shall include identification, visual inspection for damage, functional test, as required, and compliance to purchase order requirements. The contractor shall report to the government representative any nonconformance found on government source-inspected supplies and shall require his supplier to coordinate with his government representative for corrective action.

b) Production Processing and Fabrication - For new items, in-house factory fabrication, assembly and test operations shall be performed to Quality approved fabrication plans, rework instructions, and test procedures which include required inspection check points. Process control procedures shall be an integral part of the inspection system when such inspections are required by the specification or the contract.

c) Completed Item Inspection and Testing - Through review of test procedures and fabrication planning, Quality shall assure final inspection and test of completed products. Such testing shall be in accordance with the test requirements of the applicable end item specifications. When modifications, repairs, or replacements are required after final inspection or testing, there shall be reinspection and retesting of any characteristics affected.

d) Handling, Storage and Delivery - Quality shall assure that work and inspection instructions are provided and utilized for handling, storage, preservation, packaging and shipping to protect the quality of products and prevent damage, loss, deterioration, degradation, or substitution of products.

e) Nonconforming Material - The system in use for control of nonconforming materials and supplies shall conform to USAF Specification Bulletin NR-515. This system assures rapid identification, reporting and formal documentation of nonconforming articles and includes: (a) segregation to the extent required; (b) initial review and disposition by Quality; and (c) Material Review Board (MRB) action by certified MRB members when required.

f) Indication of Inspection Status - The inspection stamp control system presently in use by Martin Marietta, and which will be used for this program, provides for the identification, issuance and control of stamps, and traceability of individual stamps to the authorized custodian. These inspection stamps in use by Martin Marietta do not resemble government stamps.

g) Article and Material Controls - The Contractor presently maintains a positive system for identifying the inspection status of its products. Accepted material, articles, and/or containers are identified with Quality acceptance stamps. Nonconforming material and articles are identified and segregated until disposition is completed. Limited life materials are identified with date-of-expiration labels and items requiring contamination control are identified with clean level tags, indicating the clean level. Identification of inspection status of products shall be evident on tags or labels attached to the hardware, routing cards, move tickets, or other normal control devices.

5) <u>Coordinated Government/Contractor Actions</u>

Quality Procurement shall assure that all purchase orders stipulate a requirement which allows the Government to inspect at source, as the Government deems necessary. Purchase requisitions shall be submitted to the Government to determine Government source requirements.

C. ORBITAL TEST PLAN

Based on the experiment criteria and guidelines presented in Section B, an orbital test plan was prepared to best demonstrate performance of the cryogenic tank/feedline model thereby verifying the DSL design. The test plan presented in this section includes: (1) experiment objectives; (2) prelaunch operations and interfaces; (3) orbital events; (4) experiment operation and control; and (5) test data.

1. Objectives and Guidelines

An orbital flight test of the DSL system required to verify its low-g performance and qualify the design for future subcritical cryogen storage applications. The main test objectives are: (1) to accomplish liquid-free vapor venting for tank pressure control; (2) to demonstrate gas-free liquid expulsions at the design flow rates; and (3) to maintain near-continuous control of the bulk cryogen during the entire orbital flight.

The orbital flight test shall be an operational test of a DSL tank and feedline system using LO_2 as the test liquid. The latter is pertinent to on-orbit propulsion systems. The tank shall be sized to provide several, or more, liquid expulsions during the 7-14 day mission, while also permitting intermittent and constant venting. The vent overboard port shall be equipped with a heating element to prevent solidification of the fluid. The tank size is also traded against mass, heat leak, cost, and the VDS criteria. The feedline, or liquid supply line shall have one or more 90° bends before the liquid is lost overboard. It shall be of such volume as to permit full-line flow of liquid.

The following operational characteristics shall be evaluated: tank loading, frequency and rate of venting, sensitivity of pressure relief control, autogenous prepressurization, autogenous pressurization during expulsion, maintenance of gas-free liquid in the feedline, and the ability of the device to provide vapor communication between the gas annulus and the bulk fluid region.

The orbital test plan shall be based upon the following general guidelines:

1) A mission duration of 7 to 14 days.

2) System pressure shall be <34.4 N/cm^2 (<50 psia) for the DSL acquisition/expulsion device.

3) The test module shall be launch-ready during pad hold periods to 48-hr without the need for topping or venting, i.e., passive mode.

4) The experiment would be passive during at least the first eight hours of the mission, or that period required to satisfy the proof flight objectives, to prevent possible interference with the Centaur and Viking Spacecraft tests. Therefore, there shall be no communication with the orbital package, no outflow or venting of liquid or vapor, and negligible slosh in the LO_2 tank. The latter shall be accomplished by loading, topping, venting and passive modes such that at launch the maximum ullage condition in the LO_2 storage tank is less than 5%.

5) The storage tank shall be thermally protected to provide an environmental heating rate of approximately 1.57 watt/m^2 (0.5 Btu/hr ft^2).

6) Pressurization shall be accomplished with warm >220°K (>400°R) autogenous pressurant from a high pressure gas storage container.

2. Prelaunch Operations

Operations involving the flight test module at Kennedy Space Center (KSC) are divided into two categories: preparations and checkouts at the Spacecraft Assembly and Encapsulation Facility (SAEF); and preparation at the launch complex. This section discusses the equipment and operations for both categories.

a. Ground Support Equipment - Minimum equipment shall be required to support the prelaunch and launch operations at KSC. The same equipment shall be used at both the SAEF and the launch complex. This equipment shall include one small mobile LO_2 dewar, one cart-mounted rack of oxygen K-bottles, one control panel, and associated plumbing.

The relatively small size of the flight test module permits loading operations to be accomplished with portable equipment. A LO_2 dewar with a capacity of 0.569 m^3 (150 gallons) is sufficient. The dewar shall be equipped with a self-pressurization coil to permit pressure transfer of the oxygen. Manual valves shall be used to control loading operations. The dewar shall be mounted on small wheels which permit it to be moved by hand.

Transfer lines used for loading LO_2 shall be vacuum jacketed. Manual couplings shall be used to connect the transfer lines to the flight test article.

The GO_2 pressurization shall be supplied from the cart-mounted rack containing eight K-bottles. The GO_2 shall be controlled with a manual regulator and control valves.

The portable control panel is used to monitor and control prelaunch operations. This panel shall be connected to the test article through an electrical disconnect on the module interface. The control panel shall provide local control of the valves on the flight test article during ground operations. In addition, the panel provides visual readouts of the LO_2 tank pressure, pressurant sphere pressure, and LO_2 tank liquid level sensors.

The ground operations shall require that gaseous nitrogen be available for line purges. This nitrogen shall be obtained from available supplies at the launch facility.

b. *SAEF Operations* - The operations at the SAEF include final system checkout, pressurant loading, mating to the VLC Adapter, and encapsulation in the Centaur shroud. After arrival of the flight test module at the SAEF, it undergoes a series of final check tests to: (1) assure that no damage resulted during shipment; and (2) verify the test module's readiness for launch. A visual inspection would be made to locate any damage to components, wiring, and insulation. An electrical continuity check would also be made. This continuity check includes the control circuits, power circuits, and instrumentation wiring. Where possible, a functional test of components would be made.

After establishing that the flight test article is flight ready, a combined systems operational test would be performed in a safe area of the SAEF. Operations to be verified during this functional test would include hold periods, pressurization, liquid outflow, and venting. All instrumentation and controls would be checked.

The portable propellant loading set would be assembled adjacent to the orbital test module outside the SAEF building. Following a gas pressure leak check of the complete fluid systems, liquid and gas loading lines would be connected. Fluids would be loaded according to the prelaunch procedure. After temperature stabilization, a boiloff test would be conducted to check the integrity of the vacuum-jacketed multilayer insulation system. The spare battery would be filled and the experiment electrically energized followed by a functional checkout of the airborne equipment. With the command receiver powered up, a complete checkout with the KSC ground station would be conducted. This operation would involve sampling data from all instrumentation channels. Commands from the ground station would also check the function of the experiment equipment used, pressurize the liquid oxygen tank, and expel fluid from the tank. At the completion of this checkout the fluids would be drained and the orbital test module would be moved into the SAEF for final cleaning and assembly on the payload stack. The VDS payload would then be encapsulated in the Centaur shroud.

c. *Launch Complex Operations* - The operations to be carried out at the launch complex would be limited to propellant and pressurant loading, holding, topping, and off-loading. These operations were planned to fit into the countdown on a non-interference basis. Figure III-9 illustrates how the flight test article prelaunch operations are scheduled with respect to the basic launch operations. A description of these operations is presented in the following paragraphs.

1) Flight Test Article Loading - The loading of the flight test article LO_2 tank and pressurization of the GO_2 sphere are accomplished in the eight hour period ending 26 hours prior to launch. The test article ground support equipment shall be brought to the level of the flight test article on the Mobile Service Tower (MST). All unnecessary personnel would be cleared from the immediate area during the loading operation.

   The LO_2 fill lines, pressurization lines, vent lines, and control wiring would be connected to the flight test module at the interface located inside the shroud access door. When it had been verified that all connections were complete, a tank purge and cooldown would be initiated. The flow of oxygen from the dewar would be throttled to a slow flow rate initially to control the cooldown and purge.

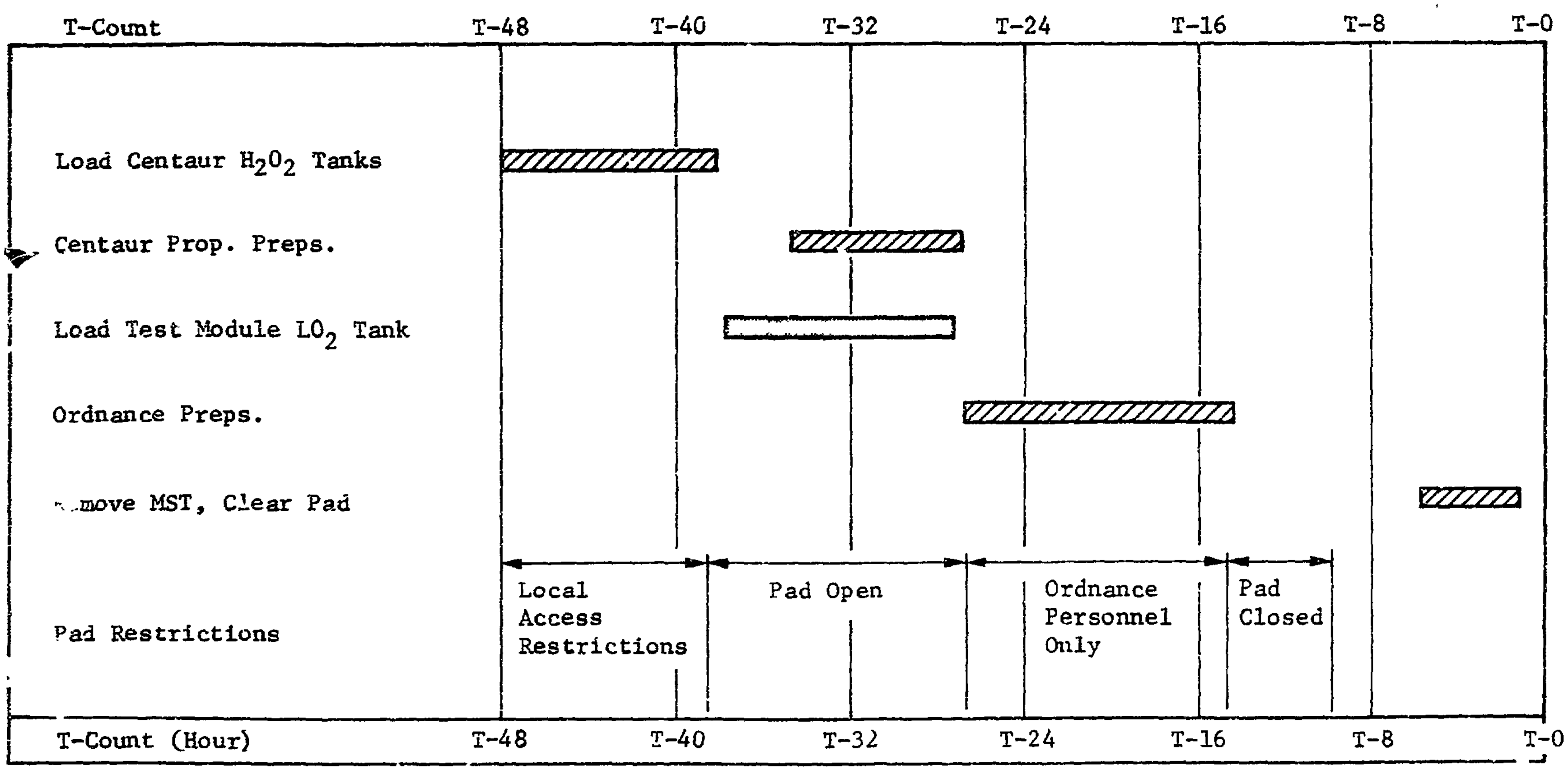

Fig. III-9 Prelaunch Timeline

When the transfer lines and tank had cooled down, the fill rate would be increased and the test article tank filled. The final 5% of fill would be accomplished at a reduced rate to permit the tank to be filled to a maximum level. When the 98% liquid level sensor was covered, the LO_2 transfer would be stopped and the tank allowed to stabilize for one hour.

The test article pressurant sphere (GO_2) would be filled during the LO_2 tank stabilization period. The sphere would be pressurized to 1720 N/cm^2 (2500 psi) and then allowed to stabilize. After both the LO_2 tank and pressurant sphere had stabilized, they would be topped as necessary to bring them to a full load.

When the LO_2 tank and GO_2 pressure sphere are stable with their correct loads, ground connections are broken and all connections capped. The nonpropulsive discharge nozzle (tee-shape) would be connected to the LO_2 feedline outlet at this time.

2) Launch Pad Hold - The design of the flight test article permits it to remain unattended from completion of loading until launch. Figure III-9 shows there would be no access to the umbilical tower during the final twenty-four hours of the count.

The design analysis presented in Section D showed that if insulation of the LO_2 tank and feedline was multilayer insulation (MLI), the LO_2 storage tank could hold for 52 hours and lose only 2% of the LO_2 through venting. Further hold time, without any venting, could be obtained by allowing the LO_2 tank pressure to rise. If the pressure were allowed to rise to 24.1 N/cm^2 (35 psia), an additional hold time of 53 hours could be obtained. Thus, by combining periods of venting and non-venting, the hold could be extended to a total of 105 hours, or approximately 4-1/4 days. This is longer than the Centaur recycle time. Therefore, holds exceeding the Centaur stage capability could be achieved with the flight test article.

This was a conservative analysis based on an assumption that the total heat leak would be into the LO_2 in the vapor annulus. Actually, there would be some mixing of the bulk liquid region with the DSL dodecasphere. Any mixing of the warm LO_2 in the annulus with the inner region would tend to extend the pad hold times.

3) Launch Abort Procedure - The long hold capability of the flight test article also eases the problem of launch abort procedures. In all but the most extreme emergency, the flight test article could remain unattended until the MST could be returned to the launch pad. The propellant and pressurant could then be off-loaded in a normal manner.

Should an emergency make it necessary to off-load without access to the flight test article, the GO_2 could be vented through the flight vent and the LO_2 dumped through the flight discharge port. The location of the port would permit the LO_2 to be vaporized and removed from the shroud by the shroud air conditioning system. The weight flow of nitrogen from this system would be sufficient to prevent the buildup of hazardous concentrations of GO_2.

3. Orbital Timeline of Events

A discussion of the orbital flight test of the DSL acquisition/expulsion system is presented in the following section. The performance and operational characteristics of the storage system are also discussed along with a summary of the anticipated test results from the orbital flight.

a. *System Duty Cycle* - The test sequence to be followed during the orbital test includes those events typically performed during an engine duty cycle for a cryogenic, earth-orbiting vehicle, such as an orbiter or Space Tug. During the first eight hours, or more, of the proof flight, launch-to-Centaur main engine cutoff, the DSL experiment remains passive. The Centaur mission sequence was described in Section A.

The following events would be performed by the test module during the mission: establishment of a vapor region in the outer annulus; low-g coast with venting; autogenous prepressurization prior to liquid expulsion; pressurization and liquid expulsion; pressure collapse following expulsion; low-g coast without venting; and liquid outflow to depletion.

A timeline of these events for a representative duty cycle is presented in Table III-7. This timeline of representative flight conditions yields a realistic appraisal of the operational performance of the DSL passive propellant control device. This postulated timeline provides for 10 separate liquid expulsions which are representative of ΔV and RCS demands on the acquisition/expulsion device. The liquid expulsions would be of equal duration, 140 sec, with an outflow rate of 0.113 kg/sec (0.25 lbm/sec). The end of the feedline would be configured such that the liquid would be expelled overboard in a non-propulsive manner.

Table III-7 Experiment Timeline of Events

Event	Mission Time (hr:min:sec)	Feedline State	Event Duration (sec)	LO_2 Mass (lb_m)	LO_2 Mass (kg)	Telemetry Trans. Time for Event (sec)
Launch	0:00:00	Dry	00	546	248	00
Expulsion to Empty Vapor Annulus	8:00:00	Wet	600	396	180	3600
Coast No Venting	8:10:00	Wet	3000	396	180	-
Pre-Pressurization	9:00:00	Wet	20	396	180	-
ΔV Expulsion	9:00:20	Wet	140	361	164	3600
Coast with Venting	9:02:40	Wet	36000	361	164	-
Pre-Pressurization	19:02:40	Wet	120	361	164	4500
ΔV Expulsion	19:04:40	Wet	140	326	148	-
Coast with Venting	19:07:00	Wet	36000	326	148	900
Pre-Pressurization	29:07:00	Wet	120	326	148	4500
ΔV Expulsion	29:09:00	Wet	140	291	132	-
Coast with Venting	29:11:20	Wet	72000	291	132	1800
Pre-Pressurization	49:11:20	Dry	120	291	132	4500
ΔV Expulsion	49:13:20	Wet	140	256	116	-
Coast with Venting	49:15:40	Wet	72000	256	116	1800
Pre-Pressurization	69:15:40	Wet	180	256	116	4500
ΔV Expulsion	69:18:40	Wet	140	221	101	-
Coast with Venting	69:21:00	Wet	36000	221	101	900
Pre-Pressurization	79:21:00	Wet	180	221	101	4500

Table III-7 (concl)

Event	Mission Time (hr:min:sec)	Feedline State	Event Duration (Sec)	LO_2 Mass (lb_m)	LO_2 Mass (kg)	Telemetry Trans. Time for Event (sec)
ΔV Expulsion	79:24:00	Wet	140	186	84.5	-
Coast with Venting	79:26:20	Wet	36000	186	84.5	900
Pre-Pressurization	89:26:20	Wet	240	186	84.5	4500
ΔV Expulsion	89:30:20	Wet	140	151	68.6	-
Coast with Venting	89:32:40	Wet	72000	151	68.6	1800
Pre-Pressurization	109:32:40	Dry	240	151	68.6	4500
ΔV Expulsion	109:36:40	Wet	140	116	52.7	-
Coast with Venting	109:38:00	Wet	72000	116	52.7	1800
Pre-Pressurization	129:39:00	Wet	240	116	52.7	4500
ΔV Expulsion	129:43:00	Wet	140	81	36.8	-
Coast with No Venting	129:45:00	Wet	72000	81	36.8	1800
Pre-Pressurization	149:45:20	Wet	180	81	36.8	4500
ΔV Expulsion	149:48:20	Wet	140	46	20.4	-
Coast with No Venting	149:50:40	Wet	144000	46	20.9	-

Liquid expulsion during the orbital test would be accomplished both with the screen feedline initially dry (no LO_2 present) as well as filled with LO_2. Following tank fill on the pad, the valve connecting the screen core portion of the feedline and the manifold at the tank outlet would be closed. The valve connecting the outer annulus region of the tank with the outer annulus surrounding the screen core in the feedline would also be closed. The valve at the end of the feedline would be opened, allowing vaporization of the residual liquid in the screen core. This gaseous oxygen would be forced from the feedline as pressure increased, and cleared from the payload shroud by the air conditioning system. The feedline would be maintained 'dry' until the terminal orbit was achieved.

Following the third and seventh liquid expulsions, the two valves at the tank outlet would be similarly closed and the valve at the end of the feedline opened, allowing the contents of the feedline to be vented overboard (to vacuum). This 'dry' feedline condition would persist until the next liquid expulsion when the outflow valve was opened and liquid was expelled. When liquid enters a 'dry' feedline of this type, some liquid would tend to pass through the screen core into the annular region surrounding the core until the screen became completely wet and the pressure in the outer region equaled or exceeded the liquid pressure inside the core.

An initial LO_2 expulsion would be made following final orbit insertion to establish a vapor region in the outer annulus. With the 4.45 cm (1.75 in.) annular gap and the tank loaded to a 5% (maximum) initial ullage condition (assuming all ullage would be located in the annular region), approximately 68 kg (150 lb_m) of LO_2 would be present in the annular region. The vibrational frequency requirements dictated that the outer annular (vapor) region be filled with this quantity of liquid to prevent excessive slosh during the proof flight test. This quantity of liquid must be expelled to establish an all-vapor condition in the annular region and represents the nominal on-orbit operating configuration for the DSL system. Venting of the tank would be accomplished from this annular region and thus required complete liquid removal prior to initiation of liquid-free vapor venting.

With a liquid outflow rate of 0.113 kg/sec (0.25 lb_m/sec) at a tank working pressure of 27.6 N/cm^2 (40 psia), 600 sec would be required to expel 68 kg (150 lb_m) of LO_2 from the tank. To accomplish this expulsion, autogenous pressurant is introduced into the outer annulus through a diffuser, which prevents direct impingement of the relatively warm GO_2 pressurant on the wetted communication screen or channel assemblies. As liquid is withdrawn from the tank, the gas-free liquid feed channels are refilled from either the bulk liquid or the liquid in the outer annulus, depending on which source sees the greater resistance to flow. As liquid is supplied from one of the regions, the flow area common to that region and the liquid flow channel decreases, causing an increase in the flow loss through the screen while maintaining a fixed outflow rate. When this flow resistance exceeds that of liquid flow from the second liquid supply region, continued replacement will tend to emanate from this second region.

The communication screen will limit the pressure differential between the bulk region and the outer annulus by allowing the passage of pressurant gas through the wetted screen and into the central region. The pressurant displaces the bulk liquid when liquid replacement to the channels occurs from this region during expulsion. When 68 kg (150 lb_m) of LO_2 has been expelled from the tank, some liquid will still remain in the outer annulus. With the tank outflow valves closed, pressurant gas continues to be introduced into the outer annulus. Liquid remaining in the outer annulus will be forced into the bulk fluid region during this pressurization rather than having GO_2 pass through the wetted communication screen. This occurs because the resistance to liquid flow, when the liquid is in contact with the screen, is considerably less than that for the passage of vapor through the wetted communication screen.

Pressure control during the duty cycle is accomplished through proper pressurization and venting. Vaporization of liquid oxygen at the surface of the screen liner tends to thermally isolate the bulk propellant by intercepting the incoming heat. This vaporization process produces an increase in pressure in the outer vapor annulus. The pressure level in the tank can be controlled by venting gaseous oxygen from the outer annulus. If the pressure is allowed to increase (no venting or insufficient venting to decrease the pressure), gas will break through the communication screen and enter the bulk fluid region. However, to support liquid within the central fluid region, the pressure in the outer annulus must remain greater than the ullage pressure in the central region. A fine pressure control on the order of 0.21 N/cm^2 (0.30 psi) is thus required to support the bulk liquid and prevent the relatively warm vapor in the outer annulus from breaking through the communication screen.

The vent control system is designed to handle the venting of vapor from the tank and feedline during the low-g orbital flight. The following vent schemes would be utilized during the orbital flight: (1) intermittent venting of vapor from the outer annulus; (2) continuous vapor venting from the outer annulus; and (3) coast without venting.

The first venting scheme, which is preferable from a thermodynamic standpoint, functions in the following manner. The pressure is allowed to increase in the outer annulus until a pressure differential is obtained which is slightly less than the pressure retention capability of the communication screen. The vent is then opened and the pressure in the outer annulus is allowed to decrease until the minimum pressure differential needed to support the hydrostatic head of the LO_2 in the low-g environment is reached. The pressure retention capability of the 250 x 1370 mesh communication screen in LO_2 is approximately 0.269 N/cm^2 (0.39 psi). Assuming a vent pressure band of 0.206 N/cm^2 (0.30 psid), sharp-edged orifice diameter of 0.101 cm (0.040-in.) yields a reasonable vent frequency and vent flow rate.

With a fixed orifice diameter, changes in input heat leak and total system pressure level could result in continuous vapor venting for some portion of the flight. This occurs when the amount of vapor vented closely corresponds to the amount vaporized at the outer screen surface due to tank heating.

The system shall also be operated in a non-vent mode for limited periods of time. In this mode the communication screen controls the pressure difference between the outer annulus and bulk fluid region. The rate of pressure rise in the tank is greatest in the non-vent situation.

A simulation of the pressure signature for that portion of the mission duty cycle following liquid removal from the outer annulus is shown in Figure III-10. This simulation was made using the DSL Cryogenic Storage Program. A detailed discussion of this program is contained in Ref III-4 and III-12. The pressures in both the outer vapor annulus and the bulk fluid region are shown, with the pressure difference between the two indicative of the pressure difference maintained across the wetted communication screen. The percentage of LO_2 remaining in the tank at any point in time is also plotted on a zero to 100% scale. A tank input heat leak of 1.57 $watt/m^2$ (0.5 $Btu/hr\ ft^2$) was assumed. An effective gas annulus gap of 4.4 cm (1.75 in.) and an effective liquid annulus gap of 1.27 cm (0.5 in.) were used.

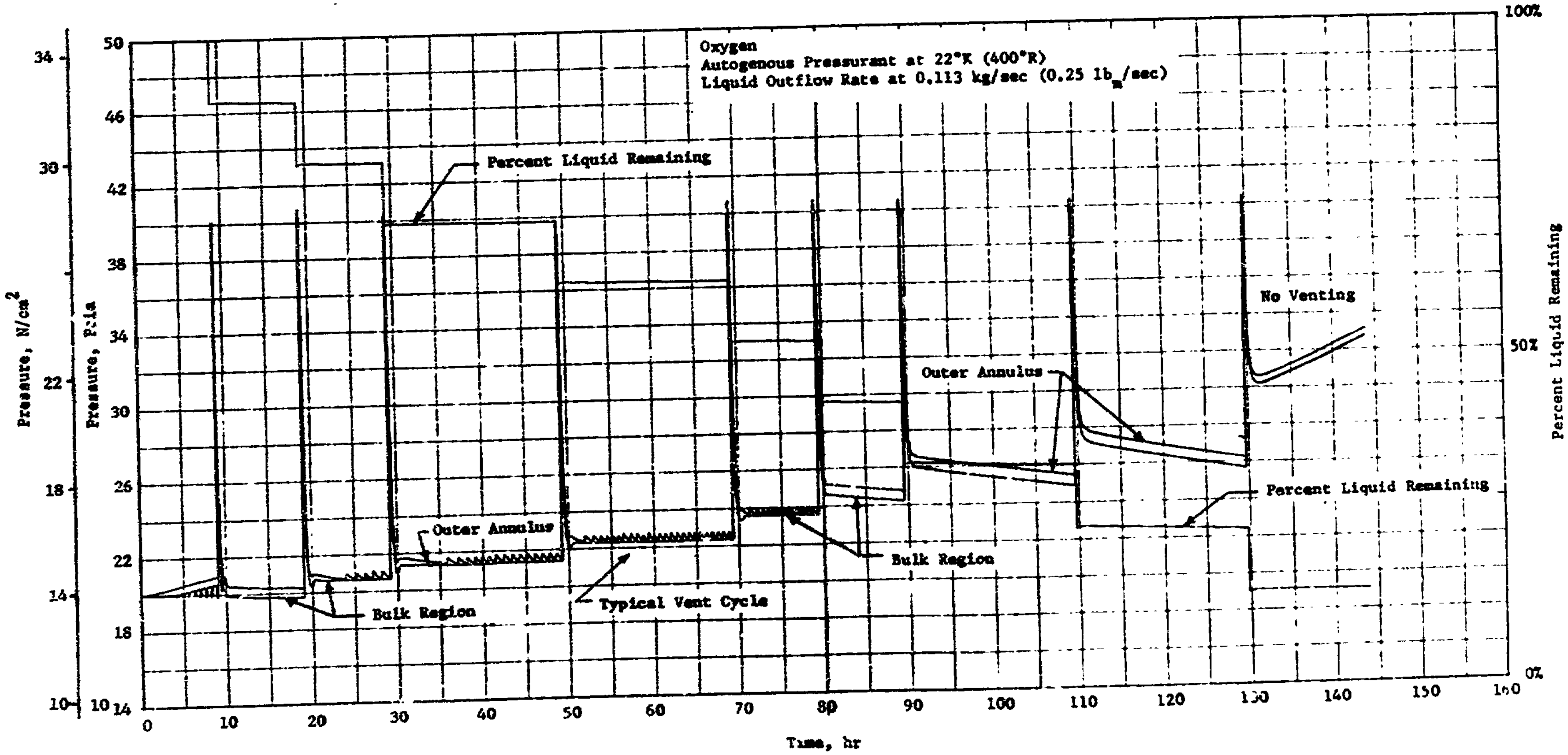

Fig. III-10 Simulation of the Orbital Experiment Mission Timeline

The pressure level in the tank during liquid outflow is controlled by a preset regulator connecting the gaseous oxygen pressurization sphere to the storage tank. Prepressurization to the tank working pressure of 27.6 N/cm^2 (psia) is performed prior to each expulsion. This pressure is maintained by the regulator during the LO_2 outflow. Between expulsions, the pressure is allowed to collapse to an equilibrium condition which results in a pressure at a slightly higher level than that following the previous LO_2 outflow. This characteristic results in a slightly smaller effective NPSP at each successive expulsion.

b. *Experiment Control* - Control of the experiment after launch would be entirely by ground command. A continuous real-time capability would exist for sending commands from the designated STDN ground stations, and for telemetered downlink data to these stations. The feasibility of this real-time transmission was explained in Section A.

4. **Experiment Performance**

Throughout the orbital test, data are obtained to verify the DSL performances. For example, the screen liner in the tank and feedline to control the bulk liquid oxygen, keeping it from the walls during low-g storage, will be verified. The DSL also stabilizes and controls the liquid in the flow channels to assure gas-free liquid expulsion. Control of the bulk liquid is necessary because it tends to minimize stratification effects, if any, and reduces propellant vaporization and resultant vented mass. In addition, data will verify that undesired vaporization in the controlled liquid region does not result during venting.

An additional stability consideration with cryogenic storage is the ability of the screen and perforated plate material to remain wetted under the imposed thermal environment. Heat leak enters the system through the tank wall and supporting structure of the capillary assembly, through soakback from the propellant feedline, and by the warm pressurization gas. Stability loss due to screen dryout is critical to the efficient operation of the system. Of particular importance is the ability of the communication screen to rewet following any dryout, thus providing the required pressure support to maintain the bulk liquid inside the screen compartment.

A measure of the system performance to supply gas-free liquid at the exit of the feedline is the expulsion efficiency. This efficiency depends on the ability of the screen forming the liquid annulus to prevent pressurization gas and propellant vapor from entering the liquid annulus and being expelled.

During expulsion, the bulk liquid is reduced and the screen area exposed to liquid flow from the bulk volume to the liquid annulus is decreased. As this occurs, the pressure drop for flow through the screen increases due to the increased flowrate per unit area. The DLS design permits nearly the entire bulk liquid to be depleted before gas is ingested into the liquid annulus and expelled. Thus, the volume within the liquid annulus is unavailable propellant. Expulsion efficiency will be measured from the terminal draining data collected.

5. Operational Characteristics

A matrix of operational characteristics for the DSL acquisition/expulsion system was presented in Table II-2. The marks indicate those parameters that are critical to the performance during the orbital flight. High-g (boost), as well as low-g, performance is considered, with the multiple sequence section indicative of system operation during the major portion of the flight. The initial low-g operation differs only in that capillary retention in the feedline is not required until the first liquid expulsion has been made, since the feedline is "dry" to this point in the test. As noted in the table, capillary retention in the storage tank is provided during the entire operation, including the high-g boost.

Vapor entrapment in the liquid flow channels may occur during tank filling due to wicking of the screen forming the channels prior to liquid fill. Since the ullage is oxygen vapor, vapor pockets may collapse completely during subsequent pressurizations and tank pressure buildup. Stability of the liquid flow channels during prelaunch and launch must also be considered. The flow channels are formed by a single layer of 325 x 2300 mesh screen which will support approximately 7.4 cm (2.9 in.) during launch (maximum 4.5g during boost). With the tank loaded to a 5% ullage condition and a gas annulus gap of 4.4 cm (1.75 in.), the liquid flow channel will not extend above the bulk propellant by more than 7.4 cm (2.9 in.) during boost.

The communication screen is designed with a lower bubble point than that forming the liquid flow channels, but is wetted by liquid from the channels. The wetted screen condition is required for support of the bulk propellant.

During prepressurization and pressurization, autogenous pressurant is introduced into the outer annulus. When the pressure difference between the gas annulus and bulk fluid region reaches the bubble point of the communication screen, gas will begin to break through the wetted screen and enter the bulk region. As long as the communication screen remains wetted, the pressure differential between the two regions will be controlled to the bubble point. The area of the communication screen is sufficient to handle the quantity of pressurant transferred to the bulk region to replace the liquid supplied to the channels during outflow. If the screen should momentarily dry out during pressurization, liquid is available from the flow channels to accomplish rewetting of the screen surfaces.

Following liquid expulsion, the pressure in the outer annulus and bulk ullage regions will collapse when the relatively warm pressurant is cooled. If pressure collapses faster in the bulk region, some additional vapor will cross the screen, maintaining the pressure differential at the operational level. If pressure collapses faster in the outer region, some small amount of liquid may "weep" through the screen, producing an increase in pressure in the outer region, and restoring the bulk fluid support.

During pressure relief, vaporization of liquid in the liquid flow channels is also a concern. The liquid being vaporized at the outer screen surface is at the saturation temperature corresponding to the vapor pressure in the outer annulus. When the gas pressure is reduced during venting, the liquid may tend to vaporize. Nucleation tends to occur at solid-liquid surfaces. As discussed in Ref. III-13, the minimum temperature excess (i.e., the temperature difference above local saturation temperature) at which nucleation first occurs is approximately 3.9°K (7°R) for LO_2.

6. Test Results

The data obtained during the orbital experiment would be used to verify system performance and identify system operational characteristics. More specifically, the data would be used to verify that the system provides gas-free liquid expulsion at the required flowrates and maintains tank pressure control under the low-g operating conditions without loss of liquid from the system.

Tank pressure can be controlled by efficient venting of saturated or superheated vapor. If the acquisition/expulsion system provides adequate fluid stability during the imposed acceleration environment, liquid would not become positioned over the vent and eventually

lost when tank pressure relief becomes necessary. In addition, the data should verify that vaporization in the controlled liquid region does not occur during venting. During pressurization of the storage tank, pressure data would indicate dry-out and re-wetting of the communication screen. Pressurization with autogenous pressurant should also tend to collapse any bubbles that may have formed in the liquid channels due to any superheating of the liquid during venting. Since autogenous pressurant would be used, the liquid temperature in the feedline must be maintained at the lowest temperature in the entire system, since the pressure throughout would be governed by the saturation pressure at the ullage bubble interface.

When data reduction of the orbital flight data had been accomplished, correlation would be made with analytical predictions from the DSL Cryogenic Storage Program. This program can simulate pressurization, venting, and liquid draining for a complete mission duty cycle. The contents of a spherical tank are divided into five nodes corresponding to the following volumes:

1) VGA - vapor volume in outer annulus;

2) VBU - bulk vapor volume;

3) VBL - bulk liquid volume;

4) VLBU - inner annulus liquid volume adjacent to the bulk ullage;

5) VL - inner annulus liquid volume adjacent to the bulk liquid.

Heat leak through the tank wall is input, and heat transfer between the various contacting liquid and gas nodes is calculated. The mass transfer at liquid-gas interfaces caused by evaporation or condensation is calculated. Boiling in the bulk liquid is calculated if the liquid becomes superheated. Pressure and nodal temperatures are calculated as a function of time by a forward-differencing technique. Since hydrostatic head and natural convection heat transfer coefficients are computed as functions of g-level, low-g conditions can be simulated. Condensation and vaporization are assumed to occur at a flat interface. Computer-plotted pressure and temperature histories are outputs of the program.

The following correlations would be made with the analytical predictions:

1) Pressure rise in the tank during coast;

2) Pressure response during intermittent vapor venting;

3) Liquid temperatures in the controlled liquid regions during coast;

4) Liquid temperature in the bulk region during the thermal transient associated with autogenous pressurization;

5) Pressure decay in the outer annulus and bulk storage regions following pressurization and outflow.

D. TEST MODULE ANALYSIS

Analysis of the flight test module was based on the experiment guidelines and orbital test plan, as discussed in previous sections. The test module is defined as the cryogenic flight test article (DSL tank and feedline system) and supporting subsystems. The basic DSL concept is presented in Section 1. A more detailed discussion f the concept is contained in Volume II. Analysis of the flight test article for the Titan IIIE/Centaur mission is presented in the second section of this part, and includes venting and pressurization requirements for the LO_2 tank and feedline. Structural analysis of the test module is presented in Section 3. Analyses of thermal control data acquisition, and power requirements are discussed in Sections 4 through 6.

1. Dual-Screen-Liner (DSL) Acquisition/Expulsion Device

The DSL was selected as the baseline passive retention/expulsion capillary concept under Contract NAS9-10480 (Ref III-3). The results of that program (Ref III-4) show the system to be attractive for a wide range of subcritical cryogen applications to provide liquid expulsion, pressure control, and near-continuous bulk fluid control. Subcritical storage of oxygen, hydrogen, methane, and nitrogen was evaluated parametrically. No tests were conducted under the NASA-JSC program; however, a cylindrical DSL tank was designed, built, and delivered to JSC in mid-71, at program completion. It was flown in the KC-135, using methanol as the test liquid, in November, 1971. Gas-free liquid was expelled.

Under this contract, detailed tank and feedline designs were made for liquid oxygen and liquid hydrogen storage systems. As reported in Volume II, point designs for an orbital maneuvering propulsion system using LO_2, and for the reusable LO_2 and LH_2 using Space Tug, were also made.

A number of ground tests were conducted, as well, as those reported in Volume III. Test results tend to verify critical operational characteristics of the DSL design; however, 1g stratification severely limited the liquid-free vapor tests using the 0.64m (25.0 in.) dia tank model. The test liquid for the diabatic tests was LH_2. These tests, along with the numerous bench tests performed to evaluate passing particular design and operational features of the DSL, were the basis for the orbital experiment module design. Instrumentation, system controls and test procedures, in particular, were directly incorporated into the orbital design.

The design incorporates the complete li ~r/channel system selected for the integrated OMS/RCS storage system. The design concept is shown in Figure III-7. This capillary system passively controls the bulk propellant during low-g. This is accomplished by the complete screen liner, which encloses all of the bulk propellant and isolates the propellant from the tank wall during the low-g storage period. The region between the liner and tank wall provides a controlled volume from which vapor can be vented to control tank pressure. A vapor-free liquid reservoir is formed by 12 separate screen channels attached to the outer screen liner and joined in a manifold arrangement over the tank outlet. Both the outer screen liner and the outer screen of the liquid flow channel are suppo~~ed by a perforated plate.

The liquid flow channels provide a continuous liquid path from the bulk liquid region to the tank outlet. To ensure this the complete screen liner (between the channels) is designed to a lower bubble point than the screen forming the channels. When pressure in the gas annulus rises (due to vaporization at screen) above the bubble point of the liner, gas will preferentially enter the bulk fluid region rather than the liquid region formed by the flow channels.

Vapor venting to control tank pressure shall occur from the outer annulus. A fine differential pressure control 0.21 N/cm^2 (0.30 psi) for LO_2 between this region and the bulk volume is required if venting is to prevent the passage of the relatively warm vapor through the liner, or communication screen, while providing continuous support of the bulk liquid. A detailed discussion of vent system sizing to accomplish this control is presented in the venting analysis section.

Pressirization gas is introduced into the outer vapor annulus through a diffuser to prevent direct impingement of the warm pressurant on the wetted communication screen. The pressurant flows through the wetted screen when the bubble point is exceeded. This gas displaces liquid from the bulk region and through the flow channels during liquid expulsion from the tank. Following expulsion, the pressure is allowed to decay to an equilibrium condition. Analysis of the pressurization system for the flight test article is contained in the following section.

The concentric screen liner within the liquid transfer line will, under low-g, position and stabilize the liquid from the wall allowing a vapor layer to form between the liner and tank wall. The vapor annulus of the feedline is connected to the outer annulus within the storage tank by a valve at the tank outlet. Vapor generated within

the feedline is vented overboard through the storage tank vent control system, as required. The valve at the tank outlet is closed during those portions of the mission when the time between burns is sufficiently long to allow dry-out of the liner (its temperature approaches the local ambient temperature) with re-chill prior to subsequent burns.

During each childown, some liquid will flow into the outer annulus and vaporize to provide pressure support of liquid within the screen when complete wetting of the liner occurs. The entire transfer line mass does not, as a result, have to be chilled since the liquid flows within the screen device. The feedline wall can remain uncooled, or warm. Subsequent vaporization in the outer annulus will raise the pressure in the feedline and also the pressure in the outer annulus of the storage tank. When the pressure in the outer annulus of the storage tank exceeds the bubble point of the communication screen, vapor will enter the bulk fluid region, raising the pressure in the storage tank to that of the feedline. A discussion of the transients associated with feedline operation is contained in the next section.

2. Flight Test Article Analysis

This section presents the pressurization and venting analyses that influenced the design of the flight test article. A brief discussion of the flow transients associated with the screen-core feedline configuration is included.

a. Venting Analysis - Vapor venting to control tank pressure occurs from the outer annulus region. Venting may be intermittent or continuous, as desired, while maintaining continuous support of the bulk propellant. The intermittent venting scheme is accomplished in the following manner. Vaporization causes the pressure to rise in the outer annulus until a pressure differential is obtained which is slightly less than the pressure retention capability of the communication screen. The vent is then opened and the pressure in the outer annulus is allowed to decrease until the minimum pressure differential needed to support the hydrostatic head of liquid in the low-g environment is reached.

The following criteria were established for the vent system:

1) the vented fluid shall be 100% vapor;

2) the vent shall be non-propulsive;

3) maximum tank pressure is 34.4 N/cm^2 (50 psia);

4) maximum vent pressure band is 0.21 N/cm^2 (0.30 psia);

5) during intermittent venting, the amount of time the vent is open should not be less than 10 sec for each vent cycle.

To relate vapor annulus gap size to vent rate, calculations of the pressure decay during venting were made (Ref III-4). The results for oxygen are summarized in Figure III-11. The vapor annulus gap size, δ, is plotted against $q\theta/\Delta P$, where q is heat flux and θ is the time to drop the annulus pressure (P) by an amount ΔP. Lines representing various ratios of vent-to-vaporization rates at the outer screen are shown.

In zero-g, allowable pressure decay during venting of the gas annulus corresponds to the pressure retention capability of the outer screen. For example, the LO_2 storage tank has a 250 x 1370 Dutch-twill communication screen with a pressure retention capability of 0.269 Newtons/cm^2 (0.39 psi). If the heat flux to the tank were 6.30 watts/m^2 (2 Btu/hr-ft^2) and it was desired to drop the pressure by 0.269 Newtons/cm^2 (0.39 psi) in a period of 39 seconds ($q\theta/\Delta P$ = 200), the δ must be 1.524 cm (0.6 in.) when $K = \dot{m}_{vent}/\dot{m}_{evap} = 2.0$, or 7.62 cm (3.0 in.) when K = 4. Therefore, for a 1.524 cm (0.6 in.) gas layer, the vent valve must be sized to modulate flow rates of about twice the boiloff rate. The plots in Figure III-11 show that the small vent pressure decay may be accomplished in a reasonable time with a practical annulus volume and vent rate.

The effect of gas annulus gap on vent frequency is shown in Figure III-12. Two annulus gap widths, 1.905 cm (0.75 in.) and 4.45 cm (1.75 in.), were considered for a 76.2 cm (30 in.) dia tank. The number of complete vent cycles that would occur in 7 days is plotted as a function of the vent flow rate during that portion of the vent cycle when the vent is open. The case considered was the bulk region full of liquid and heat input to the tank of 1.576 w/m^2 (0.5 Btu/hr ft^2). For this analysis the vented vapor was assumed saturated at 13.8 N/cm^2 (20 psia). Venting was assumed to occur directly from the vapor annulus to prevent breakdown of the communication screen. Venting was initiated when the gas annulus pressure became 0.241 N/cm^2 (0.35 psi) higher than the bulk ullage pressure. Venting was terminated when this annulus pressure was reduced to 0.0345 N/cm^2 (0.05 psi) higher than the bulk ullage pressure. As shown in previous studies (Ref III-3) an increase in vent frequency results from a decrease in annulus gap width, and therefore its volume.

Vent orifice diameters, assuming a sharp-edged orifice with a discharge coefficient of 0.6, are shown in Figure III-12 for the range of vent flow rates considered practical (reasonable response time) in a vent control system. Figure III-12 indicates that the number of vents tends to level out as the vent flow rate increases. This leveling occurs because at these high vent rates the amount of time required to reduce the pressure from 0.241 to 0.0345 N/cm^2 (0.35 to 0.05 psi) is small compared to the time for the pressure in the outer annulus to increase the same amount. Thus, the time increment of one complete vent cycle is basically determined by the pressure rise rate in the outer annulus, which is independent of the vent system. The asymptotic limit on number of vents corresponds

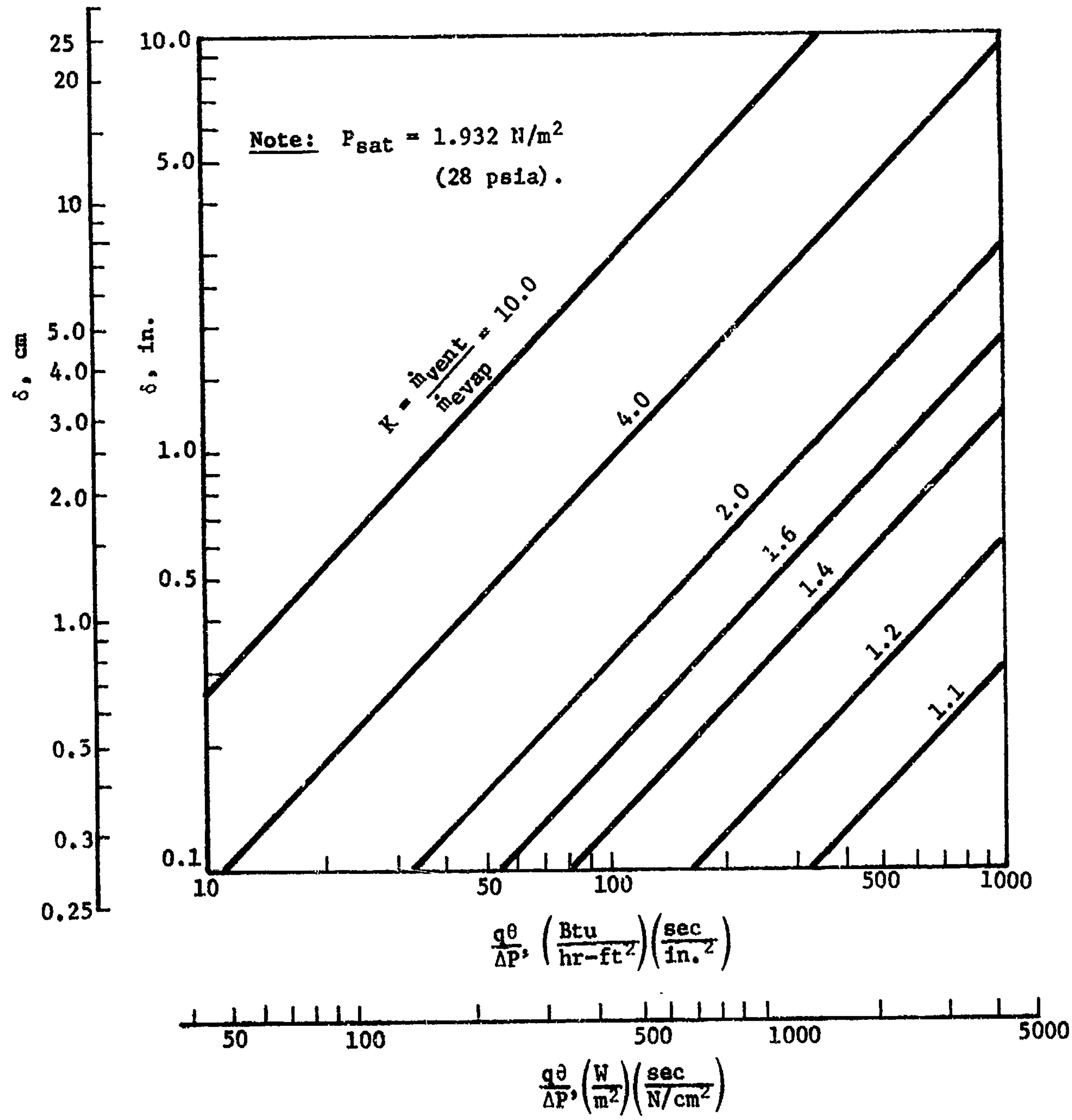

Fig. III-11 Pressu e Response during Venting

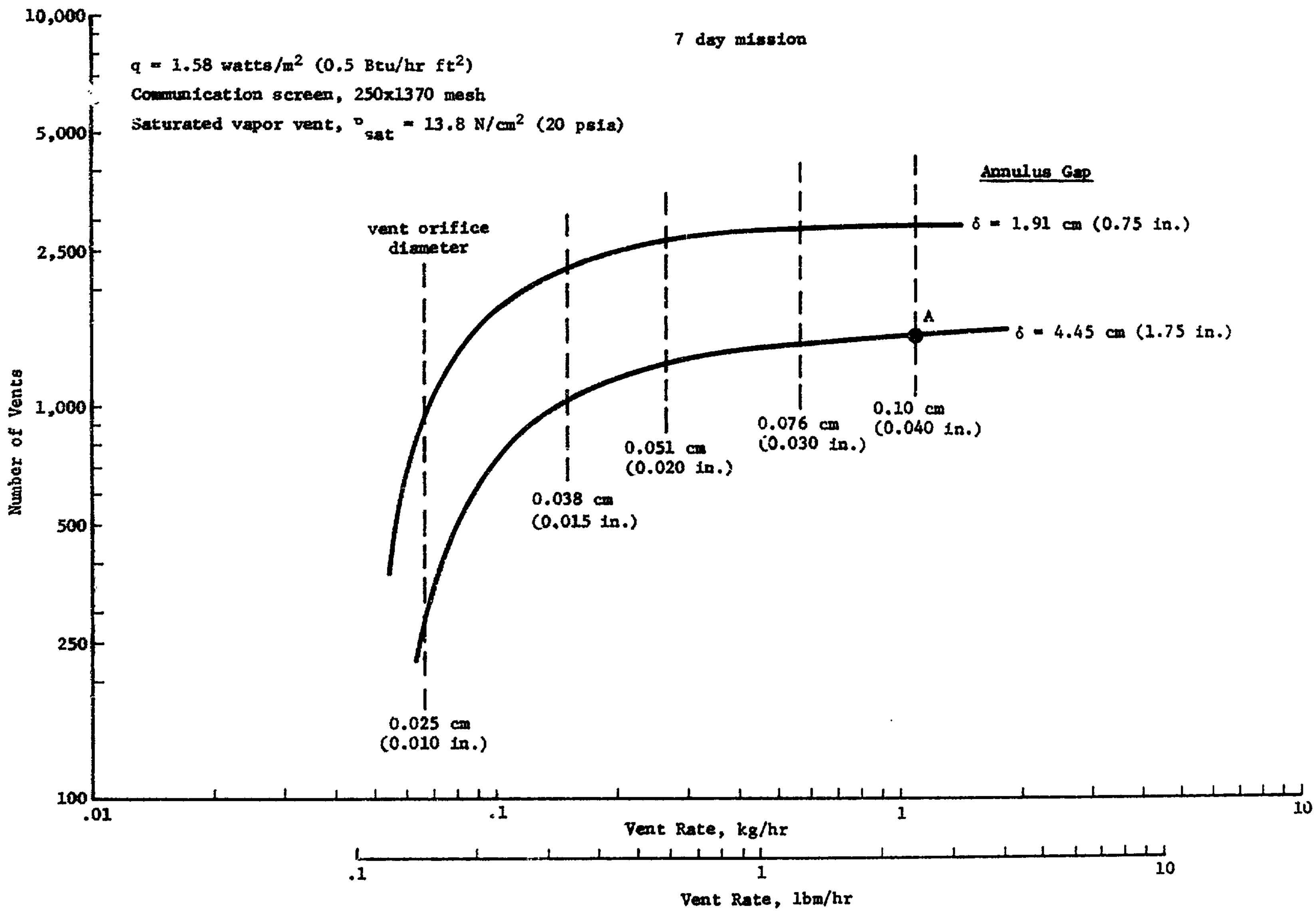

Fig. III-12 Effect of Gas Annulus Gap on Vent Frequency

to the theoretical situation where the pressure drops 0.206 N/cm^2 (0.30 psi) instantaneously when venting is initiated. When the vent orifice is sized to yield a vent flow rate approaching the LO_2 vaporization rate at the outer screen surface, a continuous vent situation is approached and the vent will tend to remain open the entire mission. This continuous venting condition requires such a small orifice diameter that plugging due to particulates or solid oxygen formation presents a reliability problem. Therefore, what was considered a reasonable orifice size of 0.1015 cm (0.040 in.) was selected for the test article. This allows a reasonable vent frequency with an annulus gap of 4.45 cm (1.75 in.).

The effect of input heat leak on vent frequency is shown in Figure III-13. As in the previous figure, the number of complete vent cycles that would occur in 7 days is plotted as a function of the vent flow rate during that portion of the vent cycle when the vent is open. The results shown are for an annulus gap of 4.45 cm (1.75 in.) in the 76.2 cm (30 in.) dia tank. Point A corresponds to the design point of the previous figure, where a 0.1015 cm (0.040 in.) orifice was selected for the vent control system. An increase in heat input results in an increased number of vents when venting occurs through a fixed orifice size. This results because the time required for the pressure rise in the outer annulus to activate the vent valve decreases with increased vaporization produced by the higher heating levels. The time required for the pressure differential to drop from 0.241 to 0.034 N/cm^2 (0.35 to 0.05 psi) (with a fixed orifice size and vapor vent rate) is independent of the heat input into the tank. Thus, as in the previous figure, the time increment of one complete vent cycle (and hence, the total number of complete vents in 7 days) is determined by the pressure rise rate in the outer annulus.

Fluctuations in vent flow rate occur with a fixed flow orifice when the operating pressure level in the tank changes. The amount of time required to drop the pressure 0.206 N/cm^2 (0.3 psi) as a function of vent flow rate is presented in Figure III-14. As the pressure level in the tank increases, the pressure drop across the orifice increases, yielding an increased vent flow rate. As the vent flow rate increases, the amount of time required to lower the pressure in the outer annulus decreases as shown. Assuming the vapor to be saturated at the orifice, the amount of time the vent is open during a complete vent cycle (vent open-to-vent open) is still greater than 10 sec at a tank operating pressure of 27.6 N/cm^2 (40 psia). With the vent orifice located remotely from the tank, it is unlikely that the vented vapor would remain saturated at the orifice. Since the orifice is thermally tied to the structure, the vapor temperature will approach the temperature of the structure, which will be controlled to a minimum value of 222°K (400°R) (see Section 4). The vent flow rates of this relatively warm vapor

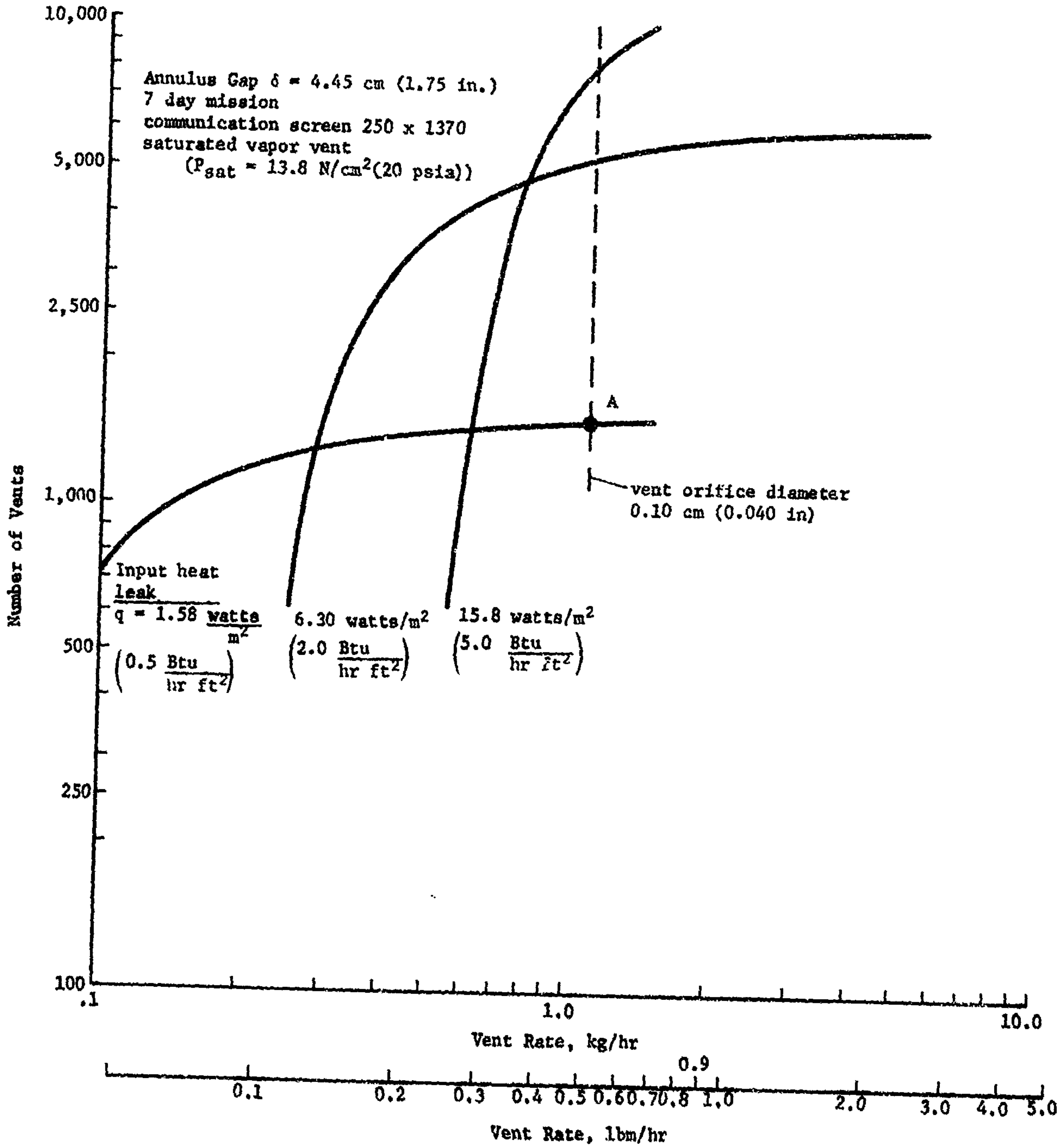

Fig. III-13 Effect of Input Heat Leak on Vent Frequency

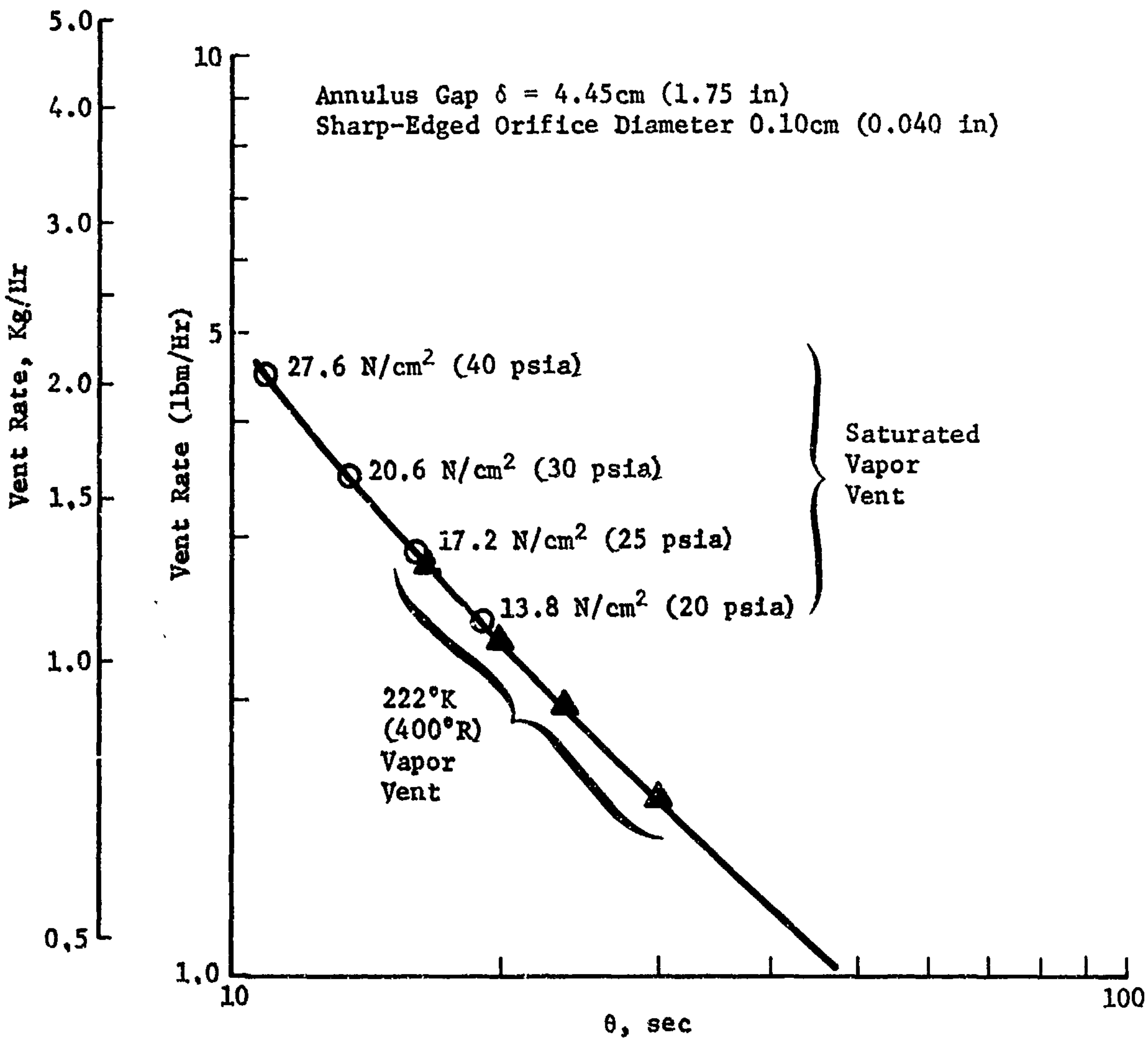

Fig. III-14 Amount of Time to Drop Pressure 0.206 N/cm² (0.30 psi) as a Function of Vent Flow Rate

through the 0.1015 cm (0.040 in.) orifice are also shown in Figure III-13. The sensitivity of vent flow rate to off-design operation is indicated in the figure. Vent flow rate fluctuations from 0.68 kg/hr (1.5 lb_m/hr) to 2.04 kg/hr (4.5 lb_m/hr) produce a total variation of 100 vent openings out of 1500 vents in 7 days. A nominal vent rate of 1.09 kg/hr (2.4 lb_m/hr), when the vent is open, results in the venting of approximately 6.82 kg (15 lb_m) of oxygen during the 7 day mission.

b. *Pressurization Analysis* - Pressurization gas is introduced into the outer vapor annulus through a diffuser to prevent the direct impingement of relatively warm gas directly on the wetted communication screen. The pressurization gas passes preferentially through the wetted screen preventing ingestion of gas bubbles into the controlled liquid flow channels. The following criteria were established for the pressurization system:

1) warm (>222°K (>400°R)) autogenous pressurant shall be used;

2) pressurant shall be diffused into the outer annulus of the storage tank during all pressurization operations;

3) prepressurization to the tank operating pressure of 27.6 N/cm^2 (40 psia) shall be performed prior to each expulsion; and

4) pressurization shall terminate immediately following each expulsion, allowing the pressure to collapse to an equilibrium condition in the tank.

The amount of pressurant required for the orbital test is dictated by the particular duty cycle to be performed. The mission duty cycle is presented in Section D. A simulation of this duty cycle was made using the Martin Marietta Corporation DSL Cryogenic Storage Program (Ref III-4). A pressure regulator was provided to control tank ullage pressure prior to, and during liquid expulsion. The total pressurant required to satisfy this duty cycle is 4 kg (8.8 lb_m) of gaseous oxygen. A 0.0283m^3 (1 ft^3) gaseous oxygen storage sphere, which contains 6.97 kg (15.32 lb_m) of GO_2 at a loaded pressure of 1720 N/cm^2 (2500 psia), was selected to provide sufficient pressurant even if increased usage occurs.

A thermodynamic and heat transfer analysis of the pressurant storage sphere was made using a typical pressurization program employing the Beattie-Bridgeman equation of state. The pressurization requirements (pressurant flowrate as a function of time), as determined from the DSL program, were input to provide the pressurant utilization from the storage sphere. The pressurant temperature in the storage tank as a function of time is shown in Figure III-15 for the 0.0283 m^3 (1 ft^3) sphere initially charged to 1720 N/cm^2 (2500 psia) at 289°K (520°R). The sphere was assumed to be insulated to prevent radiation to space. The effect of conduction in the support structures on GO_2 pressurant temperature is presented with the non-conduction case corresponding to the hypothetical situation where the sphere is perfectly isolated from the structure. With the support structure at 167°K (300°R), the pressurant temperature is maintained above 222°K (400°R) for only 23 hr into the mission. With the support structure at 222°K (400°R), the pressurant equilibrium temperature tends to approach this level a little over half-way through the mission.

The transient spikes are indicative of the temperature drop due to the cooling effects of expansion during pressurant withdrawal. As noted, the temperature during withdrawal toward the end of the mission drops below the desired level of 222°K (400 °F). Following withdrawal, heat input from the supports brings the gaseous temperature into equilibrium with the surroundings. From this analysis, it is obvious that a low conductivity interface between the storage sphere and supporting structure must be provided to assure sufficient pressurant temperature. If the pressurizing gas temperature drops below 222°K (400°R), an increased quantity of pressurant will be required to maintain the regulator-controlled pressure at the tank working pressure during liquid expulsions.

c. *Feedline Analysis* - The feedline of the orbital test module illustrated in Figure III-16 must be designed to prevent the pressure differential between the liquid-filled screen tube and the vapor annulus from exceeding the bubble point of the 325 x 2300 stainless steel Dutch-twill screen material. For liquid oxygen at nominal tank conditions this bubble point is 0.33 N/cm^2 (0.48 psid).

In a low-g environment breakdown in the feedline screen tube would be due to a combination of the friction line loss and the transient pressure pulse due to feedline valve opening or closure. For the worst case analysis of linear valve opening, the pressure drop at the valve may be expressed by Equation [III-1]:

$$P_V = P_O - P_F - P_I \qquad \text{[III-1]}$$

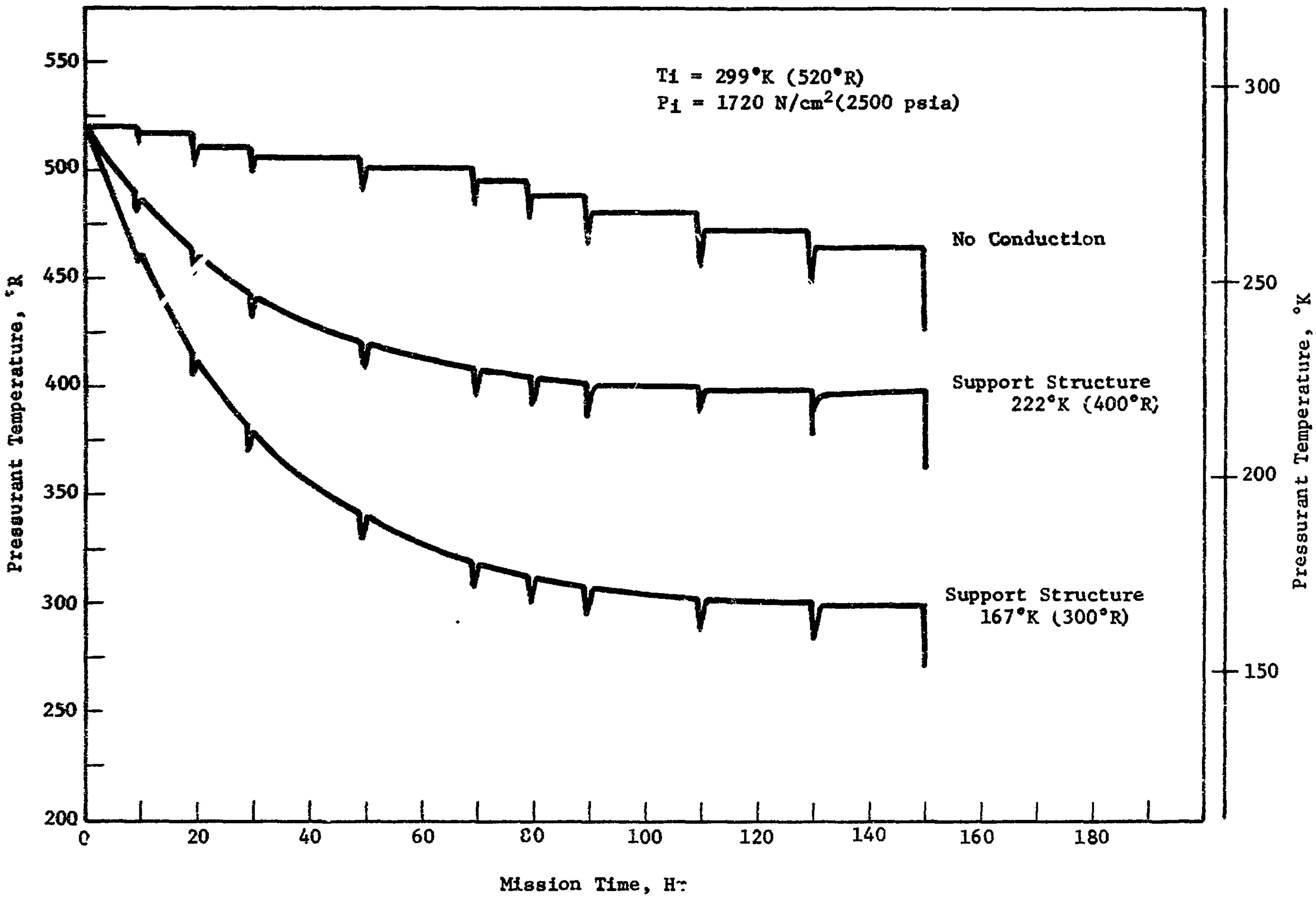

Fig. III-15 Effect Of Conduction In Support Structures On GO_2 Pressurant Temperature

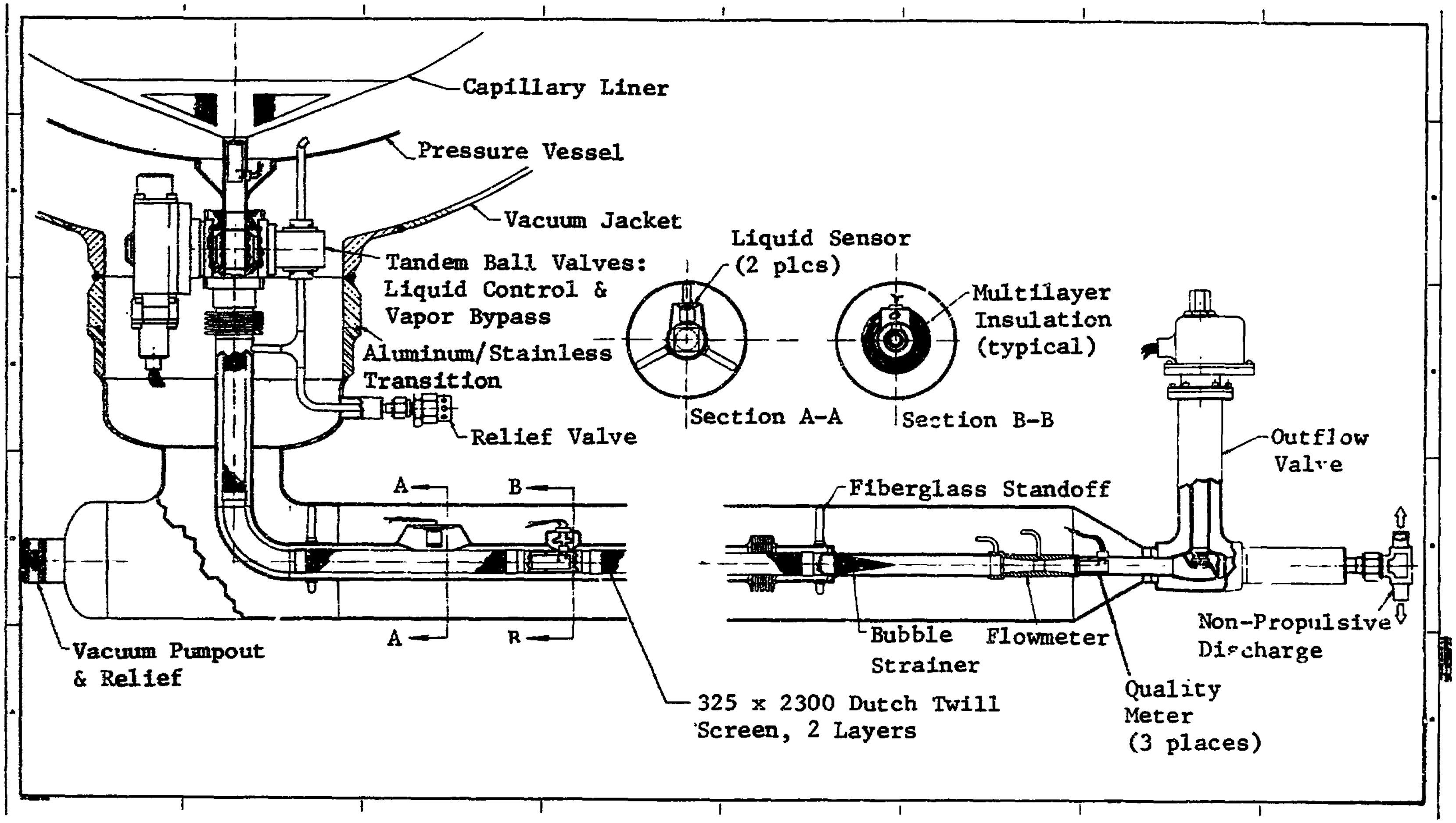

Fig. III-16 Feedline Configuration

where P_O is the tank outlet pressure, [illegible] the line friction loss and P_I the transient pressure loss due to valve opening. For linear valve opening, P_I may be expressed by Equation [III-2]:

$$P_I = \frac{2L}{A} \frac{d\dot{w}}{dt} \qquad [III-2]$$

where L is the line length, A is the cross-sectional area, and $\dot{w}$, is the steady-state mass flow rate.

The line friction loss versus mass flow rate may be determined from Figure III-17. Note that for the proposed line mass flow rate of 0.113 kg/sec (0.25 lbm/sec) the Reynold's number is apparently 3.95 x 10^4 and the friction factor, accounting for the screen roughness, is about 0.025. The total line loss is approximately 0.11 N/cm^2 (0.16 psia), well within the screen tube bubble point requirement. The value of P_I versus valve opening time for the mass flow rate of 0.113 kg/sec (0.25 lb_m/sec) is illustrated in Figure III-18.

A more refined analysis using a Martin Marietta Hydraulic Transient Computer Program (HYTRAN) is also illustrated in Figure III-18. This program may be used to model the actual valve closure characteristics, line compliance, and line friction losses. Note that the results of the refined analysis are similar to the results from Equation [III-2] for this case. Figure III-19 presents a crossplot of the maximum pressure differential at the valve versus valve opening time as determined by using the HYTRAN program. Note that the maximum pressure decompression at the valve occurs at the t = 2L/a where L is the line length and a the acoustic velocity of the transient pressure pulses. These maximum pressure excursions are rapidly damped out due to viscous losses in the line.

Figure III-19 shows that the valve opening time must be at least 0.4 sec for the pressure differential at the valve to be less than the bubble point of the screen tube material. Assuming a conservative design factor, the valve opening time should be no less than 1 sec to prevent feedline screen breakdown.

3. Module Structural Analysis

The structural system planned for the experiment test module is identical to the structure used in the GVS (Chapter IV) since dynamic similarity of the two was a basic requirement. The GVS structure was designed to meet all the load requirements of the flight module structure.

As indicated in Section A the structure was designed by dynamic requirements rather than structural loads. In order to meet the 40 Hz minimum structural frequency requirement, a very rigid and massive system was necessary. This structure, when subjected to a rigorous stress analysis, proved to be quite strong since computed stress levels were low.

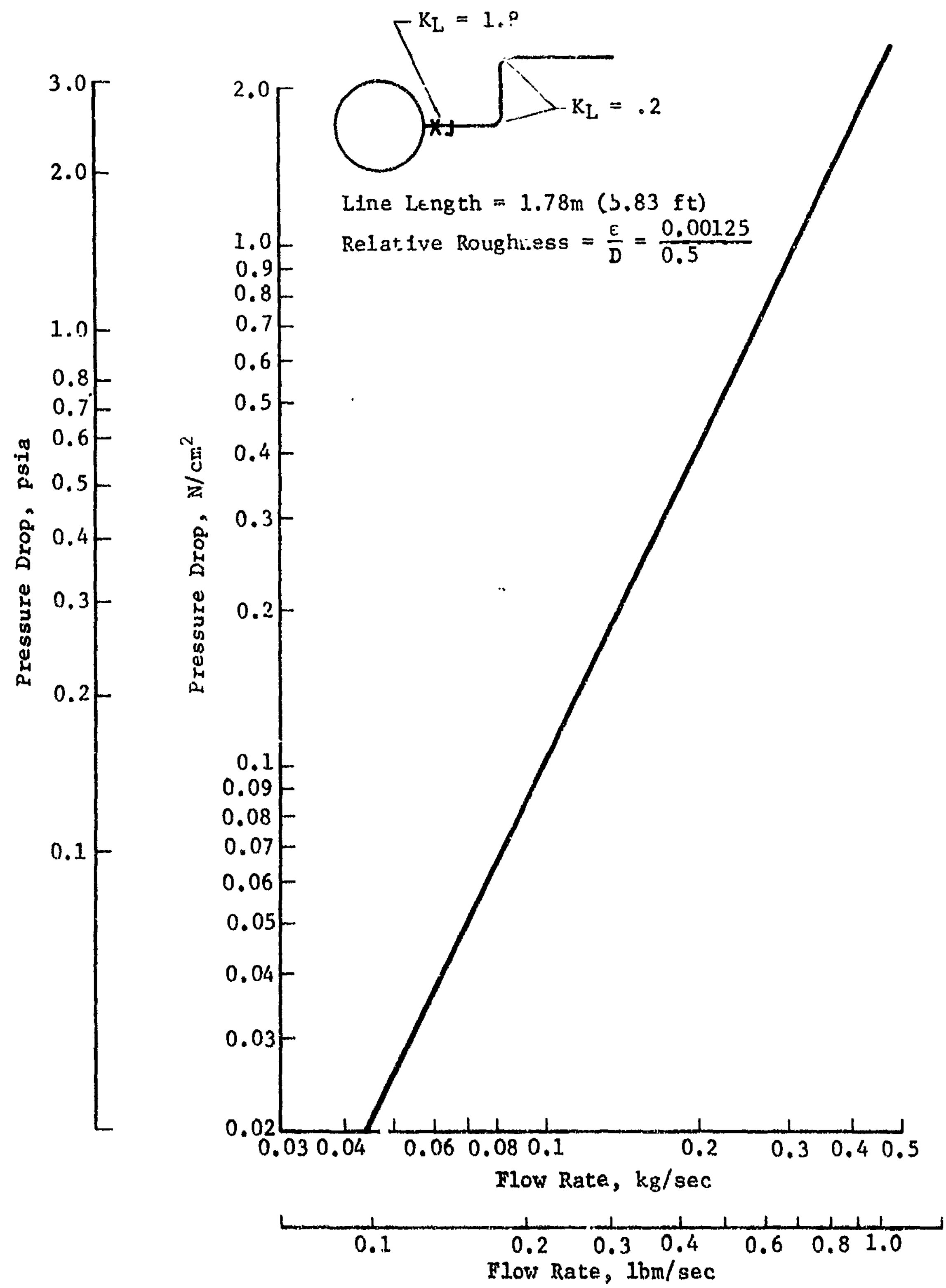

Fig. III-17 Pressure Drop for LO_2 Flow in Feedline

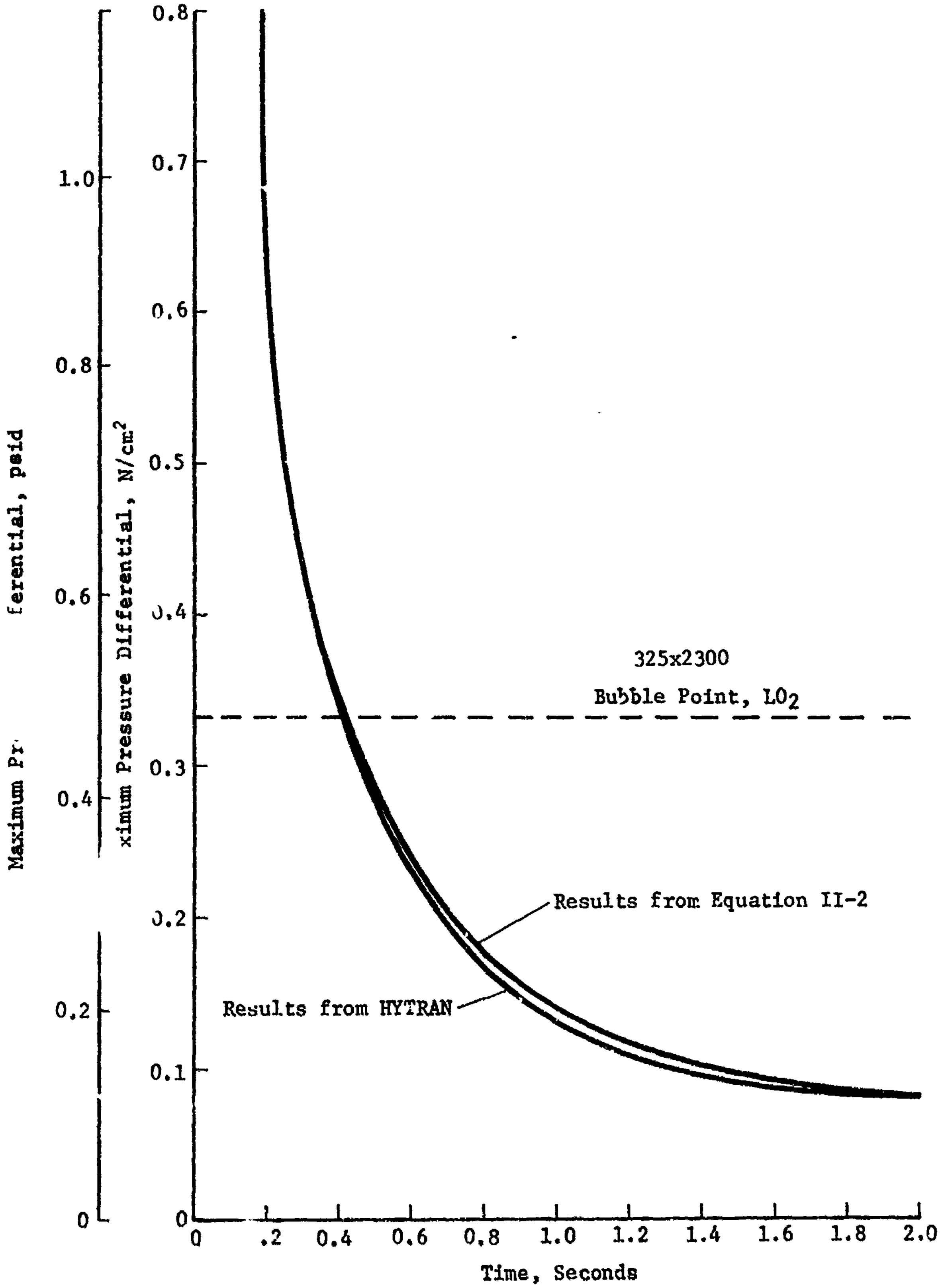

Fig. III-18 Maximum Pressure Differential vs Valve Opening Time

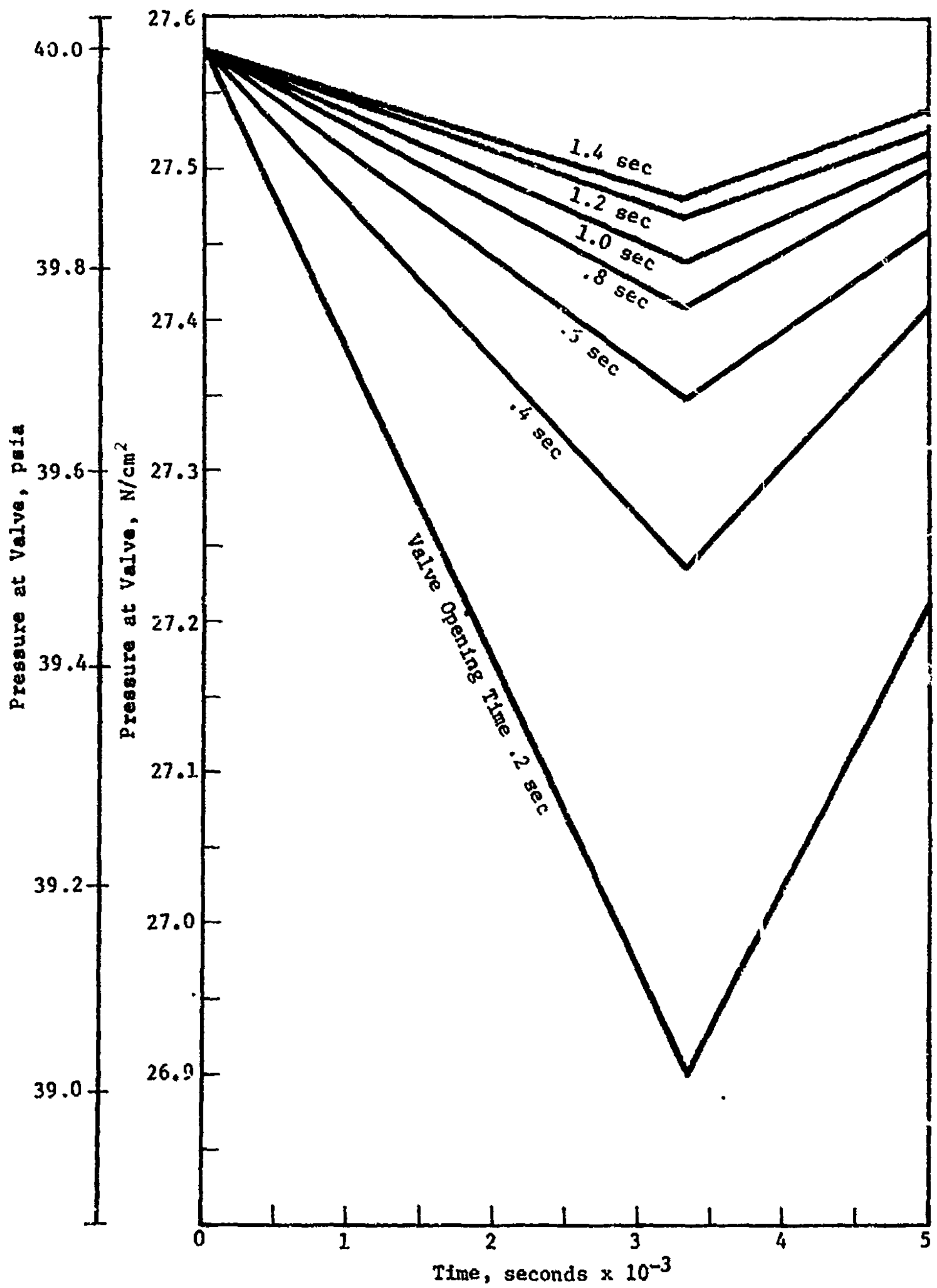

Fig. III-19 Pressure Decrease For Uniform Valve Opening Time

4. Thermal Control Analysis

a. Ground-Hold Thermal Control - Estimates of heat rate, resulting boiloff, and the nonvented pressure rise for the orbital test article were made. An ambient temperature of 295°K (530°R) was assumed for the results, as presented in Table III-8, which are appropriate for the ground-hold condition. The effective ambient temperature in orbit is likely to be relatively lower, perhaps as low as 222°K (400°R). The orbital heat rate thus is correspondingly lower, as well.

The heating of the cryogen tank can be subdivided into: (1) the uniform heat transfer through the insulation, and; (2) the concentrated heat inputs at the various insulation penetrations. The latter consist of tank supports, plumbing, and instrumentation lines. The individual contributions to the total heat load are tabulated in Table III-8. The following assumptions were made for the analysis summarized:

1) the feedline is liquid filled and vacuum jacketed;

2) the feedline valve is supported within the feedline vacuum jacket by low-conductance supports;

3) tank supports are stainless steel rods (9 places);

4) vacuum jacket annulus is 7.62 cm ([illegible] in.) thick on tank and 3.49 cm (1 3/8 in.) thick on feedline;

5) insulation thickness is: multilayer - 2.54 cm (1.0 in.); powders, foam - full vacuum space;

6) insulation thermal conductivity is:

   a) Multilayer - 1.04×10^{-6} w/cm°K (6×10^{-5} B/hr ft °R) on tank; 0.69×10^{-6} w/cm°K (4×10^{-5} B/hr ft°R) on feedline.

   b) Microspheres - 5.04×10^{-6} w/cm°K (2.9×10^{-4} B/hr ft°R).

   c) Perlite - 11.1×10^{-6} w/cm°K (6.4×10^{-4} B/hr ft°R).

   d) Foam - 130×10^{-6} w/cm°K (7.5×10^{-3} B/hr ft°R).

Table III-8 Orbital Test Article Heat Load Summary

	INSULATION TYPE			
	MLI[1]	MICROSPHERES[2]	PERLITE[2]	FOAM[2]
HEAT LOAD SOURCES	watt (Btu/hr)	watt (Btu/hr)	watt (Btu/hr)	watt (Btu/hr)
Supports	1.41 (4.8)	1.41 (4.8)	1.41 (4.8)	1.41 (4.8)
Vent Line	0.26 (0.9)	0.26 (0.9)	0.26 (0.9)	0.26 (0.9)
Instrumentation Leads	0.29 (1.0)	0.29 (1.0)	0.29 (1.0)	0.29 (1.0)
Feedline Supports	0.35 (1.2)	0.35 (1.2)	0.35 (1.2)	0.35 (1.2)
Feedline End	0.35 (1.2)	0.35 (1.2)	0.35 (1.2)	0.35 (1.2)
Tank Side Wall	1.64 (5.6)	3.02 (10.3)	6.62 (22.6)	78 (267)
Feedline Side Wall	0.65 (2.2)	1.03 (3.5)	2.26 (7.7)	26 (90)
Total Heat Load	4.95 (16.9)	6.71 (23.8)	11.54 (39.4)	107 (366)
PRESSURE RISE AFTER 24 HOURS, UNVENTED, N/cm^2 (psia)				
Stratified, with 4% Ullage[3]	14.4 (20.8)	16.4 (23.8)	22.4 (32.4)	--[4]
Well Mixed	11.5 (16.7)	11.9 (17.3)	13.2 (19.1)	--[5]
Time to Boil Off 1% of Fluid, hr	31.2	23.0	13.4	1.4

[1]One-inch thick.

[2]Full vacuum space.

[3]All heat retained in outer annulus.

[4]Reached 34.5 N/cm^2 (50 psia) in 4.25 hr.

[5]Reached 34.5 N/cm^2 (50 psia) in 14.4 hr.

The proper value of conductivity for the multilayer is a matter for discussion. For the ground-hold temperature boundary conditions, the ideal* value of the conductivity for aluminized Mylar sheets varies from about 0.14 to 0.87 x 10^{-6} w/cm°K (0.8 to 5 x 10^{-5} Btu/hr ft°R) depending on the spacer material used. The actual values achieved in practice will vary depending on the layer density achieved, the manner in which discontinuities are handled, and the degree to which the layers are perforated. For the orbital test article application, extensive perforation is not required since the insulation is not subjected to the ascent pressure profile.

The heat leaks to the tank are summarized in Table III-8. Total heat rates are shown for several insulation types.

b. *Pressure Rise* - The rate of change of internal energy with respect to pressure was computed as a function of pressure from the data of Ref III-7 is plotted in Figure III-20. When the curve is integrated, Figures III-21 and III-22 result. Figure III-21 shows the pressure which results from heating the liquid oxygen from 10.4 N/cm^2 (15 psia) as a function of heat input and ullage fraction. These curves are based on the assumption that all the energy input is retained in the outer annulus liquid.

In Figure III-22 the pressure rise as a function of heat input is shown for two initial pressures for a nearly full well-mixed tank. Figure III-21 should closely approximate (for short time periods) the ground-hold situation where boundary layer flows dominate the internal heat transfer. For long time periods, Figure III-21 overestimates the pressure rise since the diffusion of heat into the interior will tend to equalize temperatures. In orbit, the pressure rise should more closely follow the curve of Figure III-22.

The pressure attained after heating for 24 hr (initial pressure at 10.4 N/cm^2 (15 psia) are tabulated for the various insulation systems in Table III-8. These pressures are shown for both the stratified (Figure III-21) and unstratified (Figure III-22) cases. Note that in the case of the foam insulation, the pressure reaches 34.5 N/cm^2 (50 psi) in less than 24 hr. Also tabulated in Table III-8 are the times required to boil off 1% of the tank contents. Using multilayer insulation and no venting, the pressure rises only to 14.4 N/m^2 (20.8 psia) after 24 hr of heating for the stratified case or 11.5 N/cm^2 (16.7 psia) for the mixed case. With venting, 31 hr are required to boil off 1% of the LO_2 loaded.

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*Continuous unperforated parallel layers at near-optimum layer density.

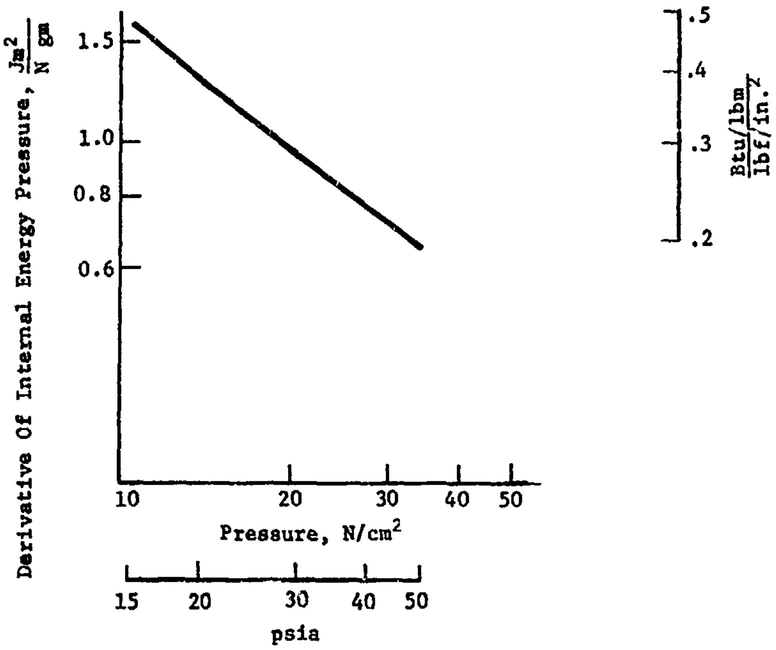


Fig. III-20 Internal Energy Function for Liquid Oxygen

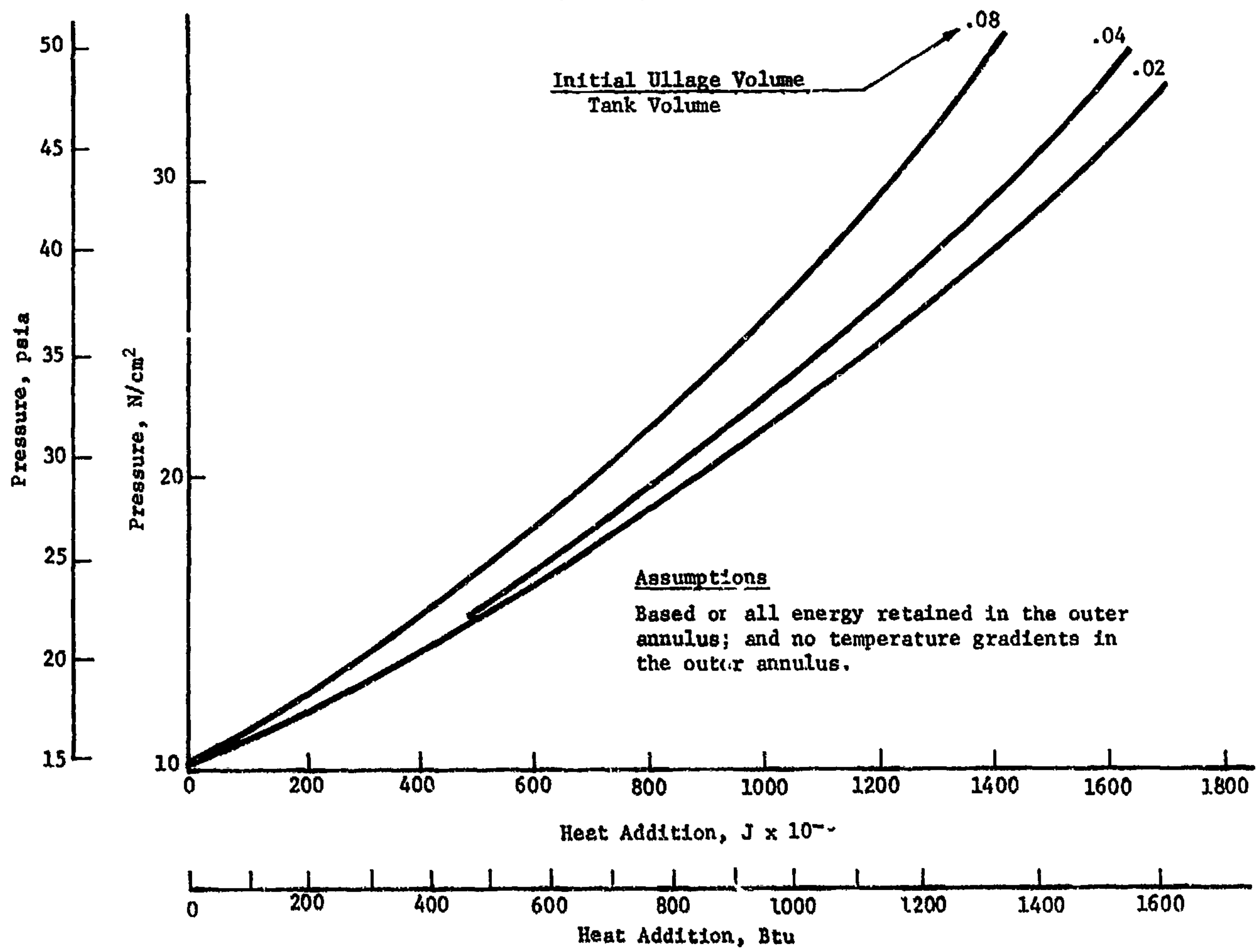


Fig. III-21 Unvented Pressure Rise in Test Article with Stratification

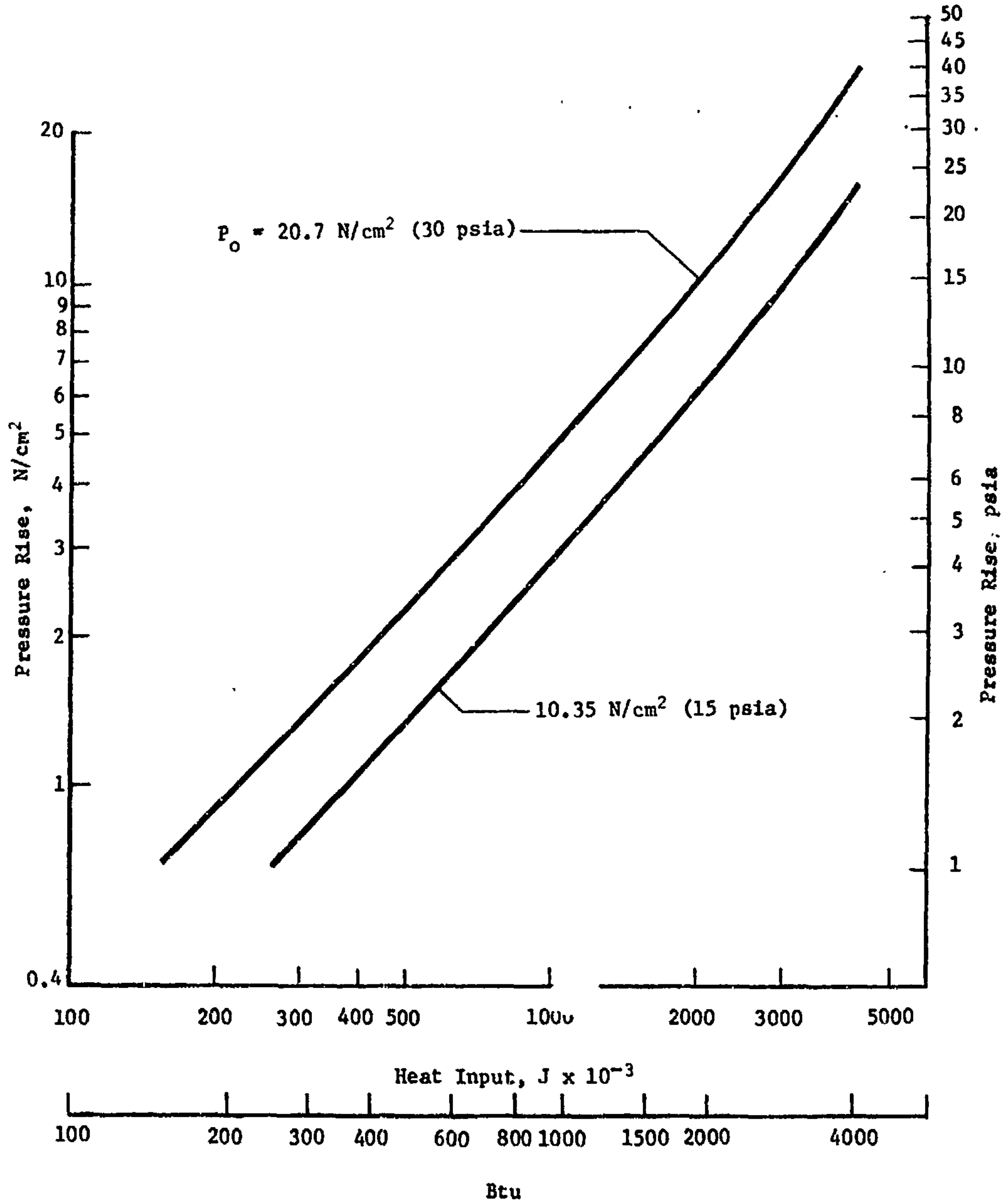


Fig III-22 Unvented Pressure Rise in Test Article with No Stratification

c. On-Orbit Thermal Control - Of all the components, the battery temperature is the most critical. The range specified is 5°C to 33°C (+40 to +90°F). The electronic packages have a temperature requirements of -17°C to 72°C (0°F to 160°F). The temperature requirements on the antennas are probably only significant in terms of temperature gradients which would lead to distortion or defocusing. The pressurization sphere temperature will be maintained above -51°C (-60°F) to insure adequate pressure. On the high side, the temperature would be limited by the allowable rise of a full tank. Assuming an allowable tank pressure of 2070 N/cm^2 (3000 psia) and an initial fill pressure of 1720 N/cm^2 (2500 psia) at 19°C (70°F) yields a maximum allowable temperature of 75°C (165°F).

The effective ambient temperature for the experiment should be sufficiently high so that the venting function can be demonstrated. A temperature around 19°C (70°F) would be ideal, though temperatures 56°C (100°F) above or below this value would be acceptable.

Both the liquid outflow and the gas vent from the experiment are controlled by orifices at the exits of the liquid and gas lines. To preclude plugging of the orifices as a result of solid oxygen formation, the orifice temperature must be maintained above the triple point of oxygen, 50°K (90°R).

The proof flight orbit is near-synchronous and elliptical with a 34,400 km (18,515 nm) apogee and 23,200 km (12,517 nm) perogee, and the inclination is 30.4°.

The experiment remains attached to the Titan-Centaur vehicle. A slow roll at the rate of two rev/hr is applied by the Centaur stage, prior to terminal orbit. The experiment will not have an attitude control capability. It would be highly desirable, therefore, as a final maneuver after orbit is achieved, that the Centaur stage roll axis be rotated normal to the solar vector. This maneuver, combined with the rotation about the roll axis, would insure a fairly uniform thermal environment for the vehicle. The maneuver would preclude the possibility of the vehicle axis aligning with the solar vector at least for the duration of the experiment. Such an alignment, if stable with the experiment in the shade, would result in uncontrollable low temperatures.

5. Data Acquisition Analysis

Requirements for measurements to be made were assembled and analyzed. Communications requirements were developed by considering the measurements to be telemetered and the command capability needed for control of all module subsystems.

a. Measurements - Table III-9 lists the quantities, categories, and types of measurements required. Experiment specialists asked for analog measurement precision to be on the order of 1%, and sampling rates of at least one sample per second for all measurements. These performance criteria focused attention on digital data systems.

For 1% precision, an analog-to-digital converter must encode samples of analog data into digital words containing at least 6 bits. Most aerospace data systems employ 8-bit encoding for analog measurements. Bilevel measurements require only one bit per sample, coded either "one" or "zero," to denote which of two possible states exists at the time the channel is sampled.

At one sample per second for each channel, and with 8-bit encoding, the 28 analog measurements listed in Table III-9 would produce a data rate of 224 bits per second (bps). The 21 bilevel measurements would produce 21 bps. With allowance for spare channel capacity and synchronization bits, the total data rate was estimated to be approximately 500 bps.

Table III-9 Data Acquisition Measurement Requirements

QUANTITY	CATEGORY	TYPE
12	Temperature	Analog
13	Liquid Level	Bilevel
4	Valve Position	Bilevel
5	Pressure	Analog
1	Flowrate	Analog
10	Housekeeping	Analog
4	Housekeeping	Bilevel

The channel quantities, sampling rates, and encoding precision became criteria for selection of data multiplexing equipment, and the data rate became one of the inputs to the telemetry link analysis described below.

b. Communications - Requirements for telemetry and command links were identified, and link analyses were performed. Before the link analyses could be started, it was necessary to conduct a trade study to select the frequency bands to be used. Factors considered in the trade study are shown in Table III-10. Emphasis was placed on minimizing program cost considering surplus hardware applicability, usability of existing component designs, and development requirements. At the conclusion of this trade study, S-band telemetry and VHF command frequencies were selected.

Maximum communication range was determined from orbit geometry and ground station constraints. A minimum ground antenna elevation angle of 5 degrees was assumed. With this angle, and with the module at the apogee altitude of 34,000 km (18,515 nm), the communication range is 38,000 km (20,500 nm). Although several STDN stations have 25.9 m (85 ft) receiving antennas, 9.1 m (30 ft) antenna was assumed so that any ground station capable of S-band reception could be used.

At S-band, the incidental frequency modulation of a state-of-the-art transmitter is comparable in magnitude to the data rate planned for this program. A spectrum-spreading technique must be applied to eliminate this problem. Use of a subcarrier, although it is not the optimum spectrum-spreading technique, was selected because it is the minimum-cost approach. The telemetry bit stream frequency-modulates a voltage-controlled oscillator (VCO), which in turn frequency-modulates the transmitter. The resulting transmitter output is identified as a PCM/FM/FM signal, with the data format being Pulse Code Modulation (PCM). Ground station compatibility is assured by selection of a standard VCO frequency.

Results of the telemetry link analysis, shown in Table III-11, indicate a need for 8.7 watts of transmitter power. Several vendors offer solid state, space-qualified S-band telemetry transmitters capable of producing this power level.

The command link analysis considered VCO transmitting capabilities at ground stations usuable for telemetry reception, and assumed the Biosat command receiver sensitivity as a representative value. A 3-dipole receiving antenna system on the module, feeding two receivers in parallel, was contemplated. The results, shown in Table III-12, indicate a link margin of 2.9 db for the greatest possible range.

Table III-10 Communications Frequency Selection Analysis

COMMAND FREQ, MHz	TELEMETRY FREQ, MHz	ADVANTAGES	DISADVANTAGES
150	136	One antenna system for TM and Command. Biosat receivers, decoders, diplexer usable. 13 compatible ground stations.	Development required for 20 watt transmitter. High receiving system temperature.
450	136	Titan 450 MHz antenna design applicable. Titan 450 MHz diplexer available.	Only 7 compatible ground stations. Development required for 20 watt transmitter. High receiving system temperature. Minimal use of Biosat hardware. Two antenna systems required.
2100	136	None identified.	Same as above
150	2250	Positive transmitting antenna gain achievable. Biosat receivers and decoders usable. Low receiving system temperature. 12 compatible ground stations.	Two antenna systems required. Antenna switching required. Low bandwidth efficiency.
450	2250	Titan 450 MHz antenna design applicable. Titan 450 MHz diplexer available. Positive transmitting antenna gain achievable. Low receiving system temperature.	New communication receivers and decoders required. Two antenna systems required. Antenna switching required. Low bandwidth efficiency.
2100	2250	One antenna system for TM and Command. Low receiving system temperature. 12 compatible ground stations. Coherent system (Unified S-Band) possible.	Low bandwidth efficiency. Expensive command receivers. Expensive transponder in USB is used.

Table III-11 Telemetry Link Analysis

Carrier margin required	+3.0 db
FM threshhold	+9.0 db
Carrier modulation loss	+2.3 db
Predetection bandwidth, 100 kHz	+50.0 db re 1 Hz
System noise density, 95°K	-178.8 dbm/Hz
Received power required	-114.5 dbm
Receiving system losses	+1.0 db
Receiving antenna gain, 30 ft	-44.0 db
Polarization loss	+3.0 db
Path loss, 2250 MHz, 38,000 km	+191.2 db
Transmitting antenna gain	0 db
Transmitting circuit losses	+2.7 db
Net circuit loss	+153.9 db
Received power required	-114.5 dbm
Net circuit loss	+153.9 db
Transmitter power required	+39.4 dbm
	(8.7 watts)

Table III-12 Command Link Analysis

Transmitter power, 5 KW		+67.0 dbm
Transmitting antenna gain, SATAN		+20.0 db
Path loss, 150 MHz, 38,000 km		-167.5 db
Polarization loss		-3.0 db
Receiving antenna gain, 90% coverage		-5.0 db
Receiving circuit losses:		
3-way combiner	-5.3 db	
2-way divider	-3.5 db	
Cable losses	-0.8 db	
		-9.6 db
Power level at receiver		-98.1 dbm
Receiver sensitivity		-101.0 dbm
Link Margin		+2.9 db

6. Power Requirements

An electrical power profile was developed to serve as the basis for battery sizing and to support the module thermal analysis. Component specifications yielded power demand data, and the mission timeline defined operating times for each component. After the power profile had been established, the maximum current requirement of 9.0 amp was determined by inspection. An energy requirement of 218.8 amp-hr was calculated by integrating the power profile over the one-week mission duration. A summary of energy requirements is presented in Table III-13.

Table III-13 Electrical Energy Requirements

LOAD	AVERAGE CURRENT, A	TIME ON, hr*	ENERGY, A-hr
Solenoid Valves	0.35	168	58.6
Transmitter	2.32	16.8	38.9
Signal Conditioner	0.67	16.8	11.2
10-Volt Power Supply	0.62	16.8	10.4
Commutator-Coder	0.21	16.8	3.5
Commutator	0.06	16.8	1.0
Relay Assy.	0.017	168	2.9
Heaters	0.53	168	89.0
Command Receivers	0.02	168	3.3
	Total Energy		218.8 A-hr

*Nominal 1-week mission.

E. TEST MODULE PRELIMINARY DESIGN

Preliminary design of the orbital test module for the Titan IIIE/Centaur proof flight mission is presented in the following sections. Design of the flight test article included the DSL screen device, LO_2 storage tank, vacuum jacket, feedline, fill line, vent system, and pressurization system. Test module subsystems included: structural support, thermal control, instrumentation, communications and data acquisition, and power. Results of the preliminary design review (PDR) conducted at Martin Marietta in December 1972 are presented at the end of this section.

1. Flight Test Article

The flight test article tank and feedline system is shown schematically in Figure III-23 Each element of the design is discussed in the following sections.

a. DSL Acquisition Device - The DSL acquisition device is contained in a 76 cm (30 in.) dia spherical stainless steel tank which is filled with 250 kg (550 lbm) of LO_2. This tank is encased in an aluminum spherical vacuum jacket 91 cm (36 in.) in dia. The acquisition device consists of a complete liner formed from 250 x 1370 mesh Dutch-twill screen and backed with perforated plate as shown in Figures III-24 and III-25. This screen liner is positioned to provide a 4.4 cm (1.75 in.) vapor annulus between tank wall and liner. Twelve flow channels, attached to the inner wall of the liner, are fabricated from a finer meshed screen than is used on the liner. This finer mesh (325 x 2300) protects the channels from vapor ingestion (Ref III-7).

Three distinct volumes may be identified in the acquisition device. These are: (1) the vapor annulus; (2) the liquid flow channels; and (3) the bulk region or the volume within the liner excluding the flow channel volume.

The complete screen liner is a 12-sided polysphere. This geometry is used to avoid compound curvature in its fabrication. Single curvature fabrication, possible with the polysphere, is preferred because compound curvature causes a significant reduction in the pressure retention of the screen. The twelve segments (gore panels) which make up the polysphere are made from 0.06 cm (0.024 in.) stainless steel sheet perforated with specific pattern of 0.63 cm (0.25 in.) holes, yielding a 30% open area. As shown in Figure III-26, the gore panels are not perforated on the edges nor in the central section. The Dutch-twill screen is attached to the panels in the unperforated areas by seam resistance weld. The edges of the panels are flanged, permitting edge weld joints between gore sections to minimize warpage.

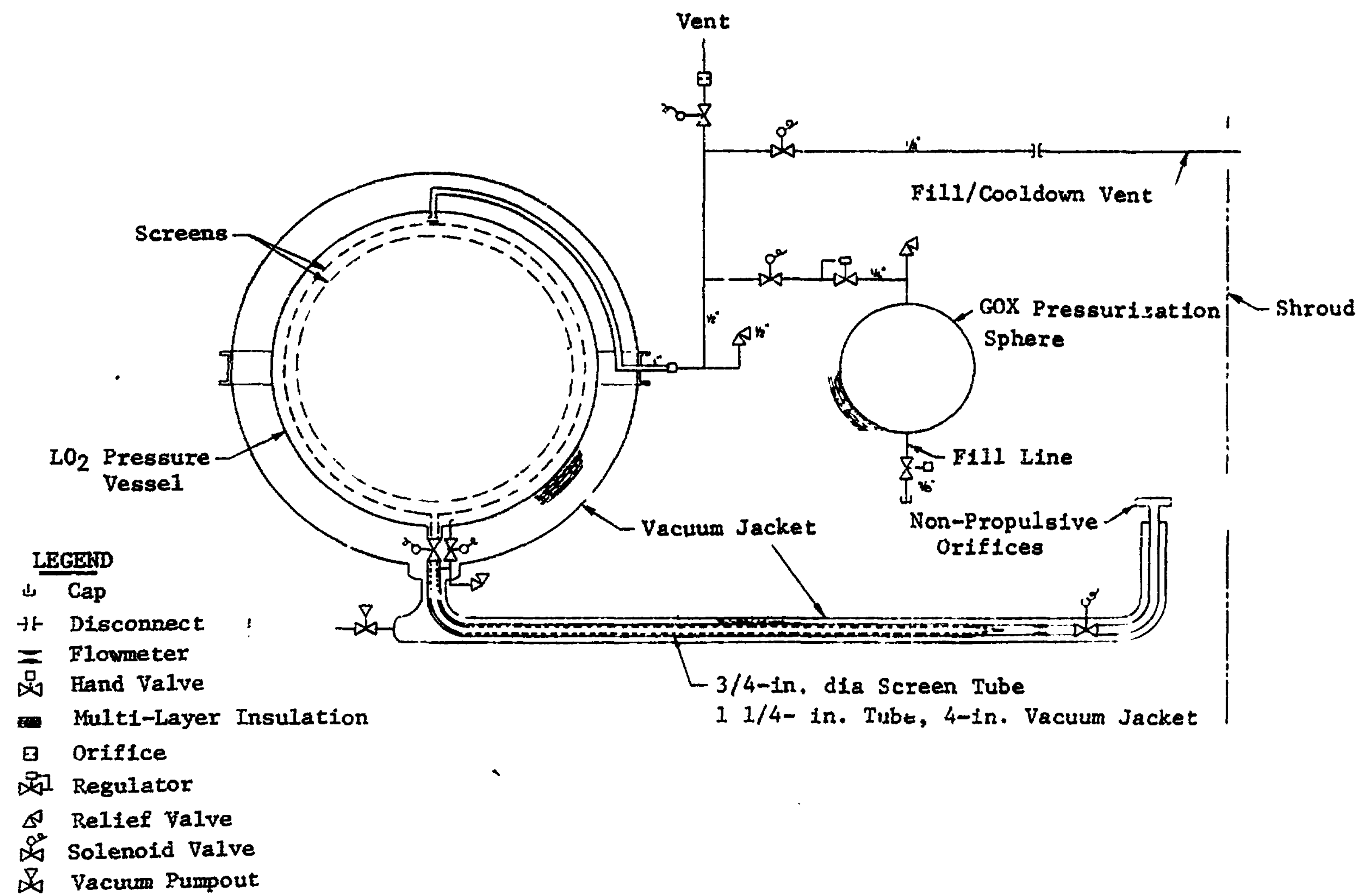


Fig. III-23 Test Article Plumbing Schematic

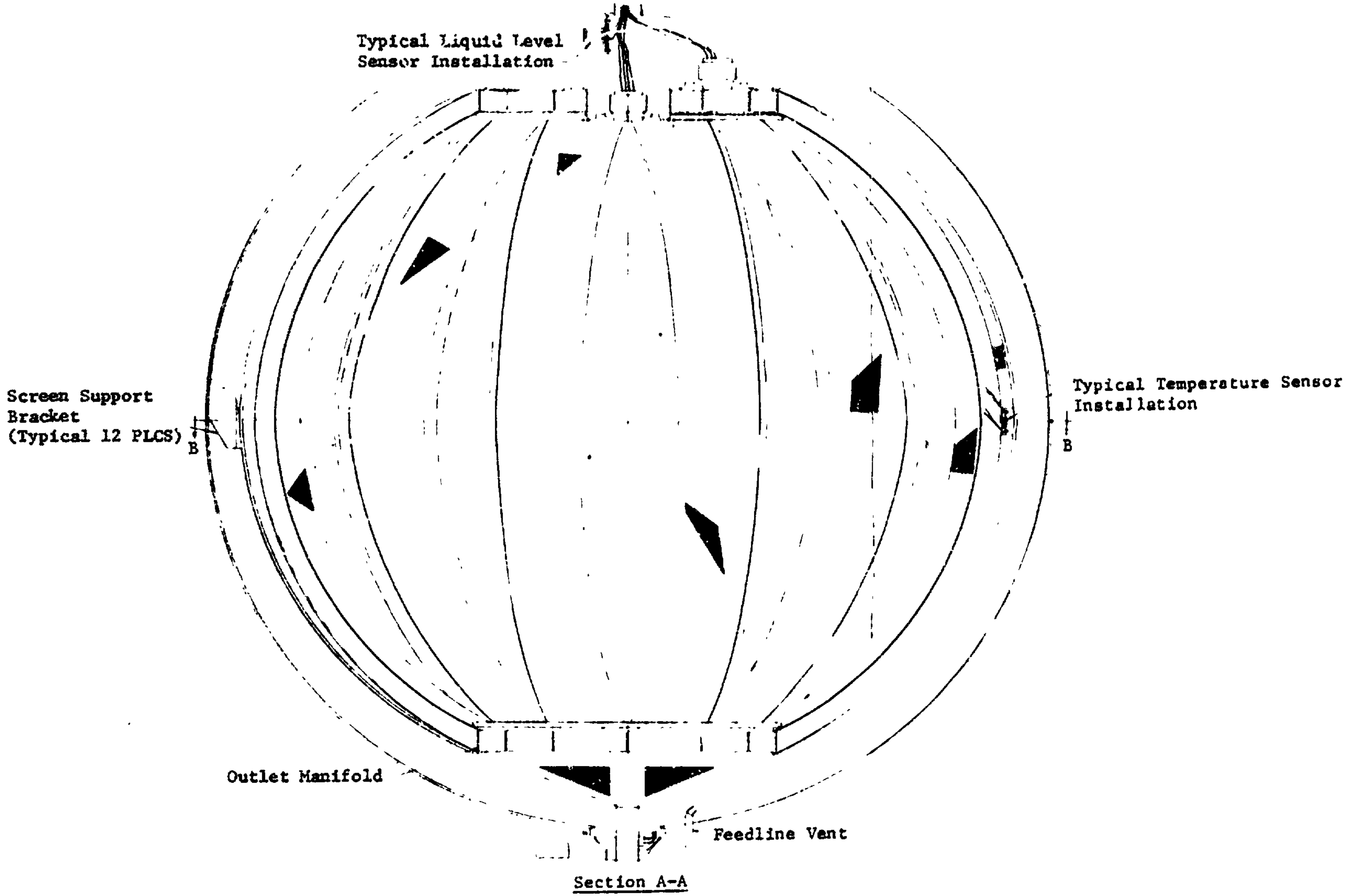
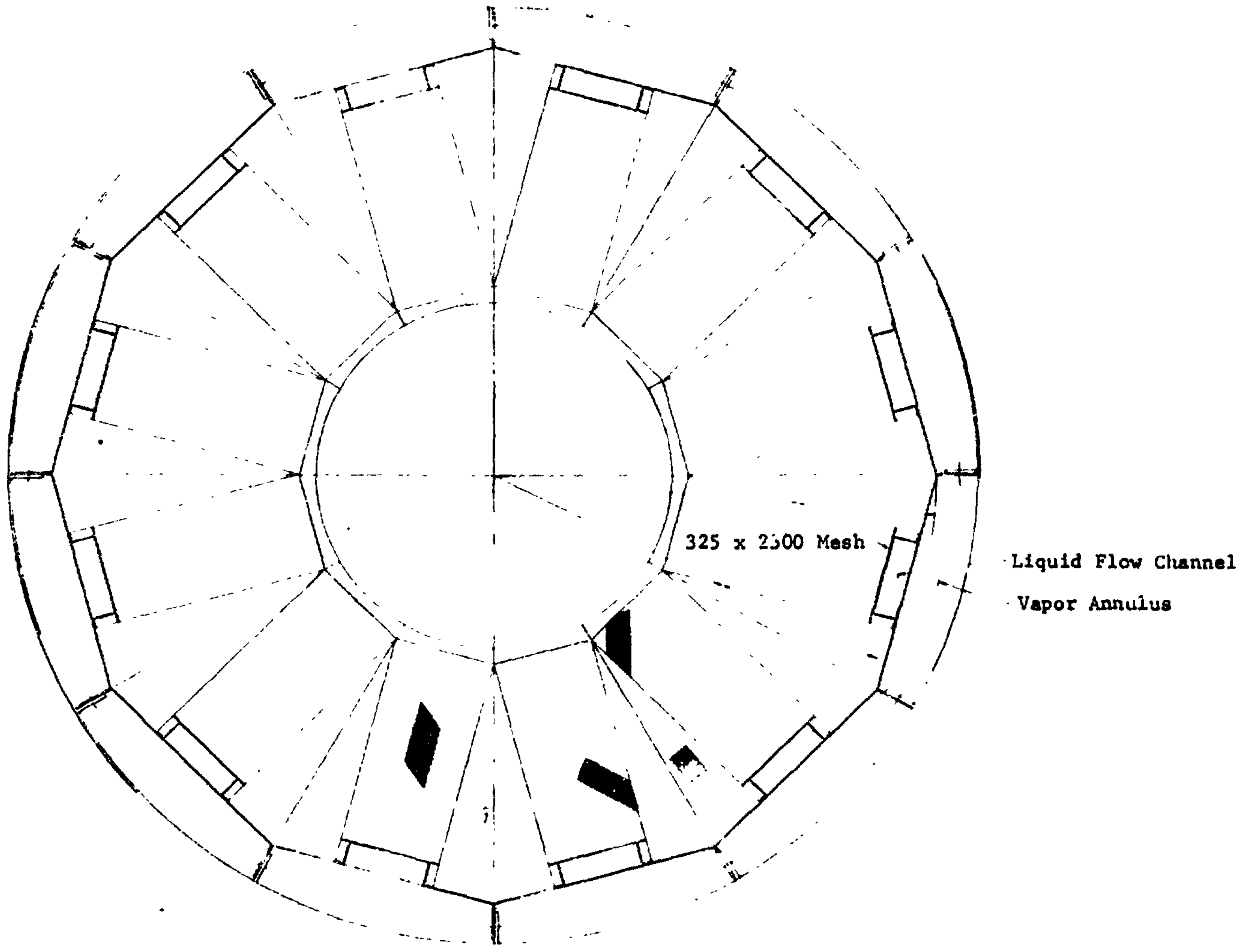
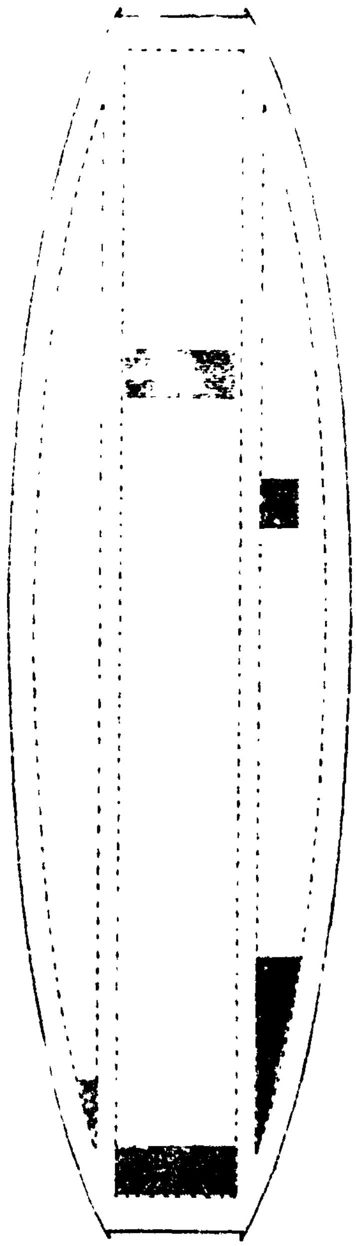


Fig. III-24 Acquisition Device Side View

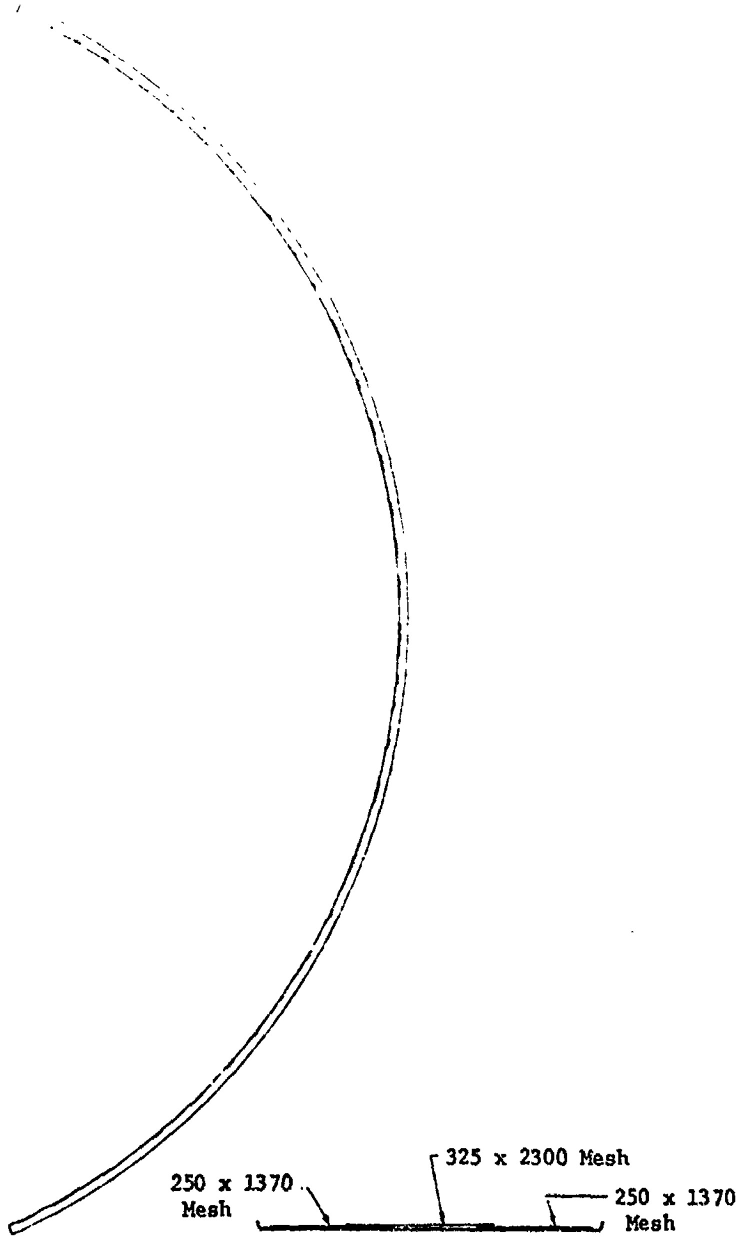


Section B-B

Fig. III-25 Acquisition Device Top View



Perforated Panel Before Screen Installation



Edge View Of Perforated Panel After Screen Installation
scale full

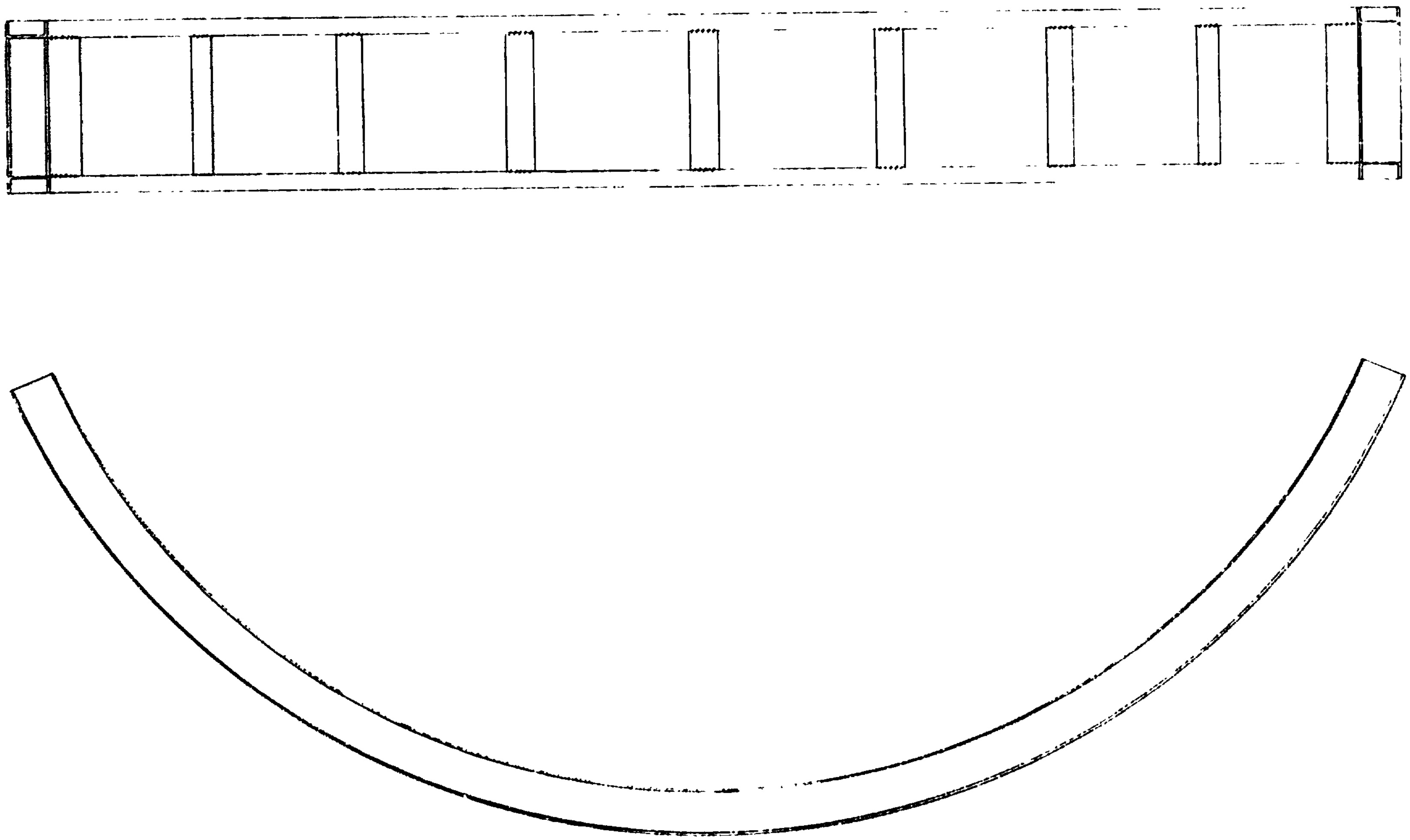
Fig. III-26 Typical Gore Panel

Flow channels of constant cross-section, 1.9 x 6.3 cm (0.75 x 2.5 in.) are contained in the polysphere liner. The channels are designed to remain full of liquid for the duration of the mission. The screens on the gore sections are entirely wetted from the liquid in the flow channels by wicking. Dutch-twill screen, 325 x 2300 mesh, is attached to both sides of constant-radius channel members formed from 0.05 cm (0.020 in.) stainless steel sheet. Screen support is provided intermittently by cross members attached to the channel sides (Fig. III-27). The flow channels are closed at the upper end and are attached to the screen covered conical manifold to jointly expel gas-free liquid to the feedline.

b. *LO_2 Storage Tank* - Two stainless steel hemispheres, spun to a 76 cm (30 in.) dia with a minimum wall thickness of 0.07 cm (0.030 in.), were selected for the storage tank. The acquisition device is supported within the tank by flanges welded to the upper hemisphere at twelve points near the girth weld. Each point is attached to a mating tab welded to the acquisition device at the junction of each gore section. The acquisition device is positioned to provide an annulus equal to a constant gap thickness of 4.4 cm (1.75 in.). The total weight of the tank and device is 36 kg (80 lbm). The loaded tank contains 250 kg (550 lbm) of LO_2 allowing for a 5% ullage.

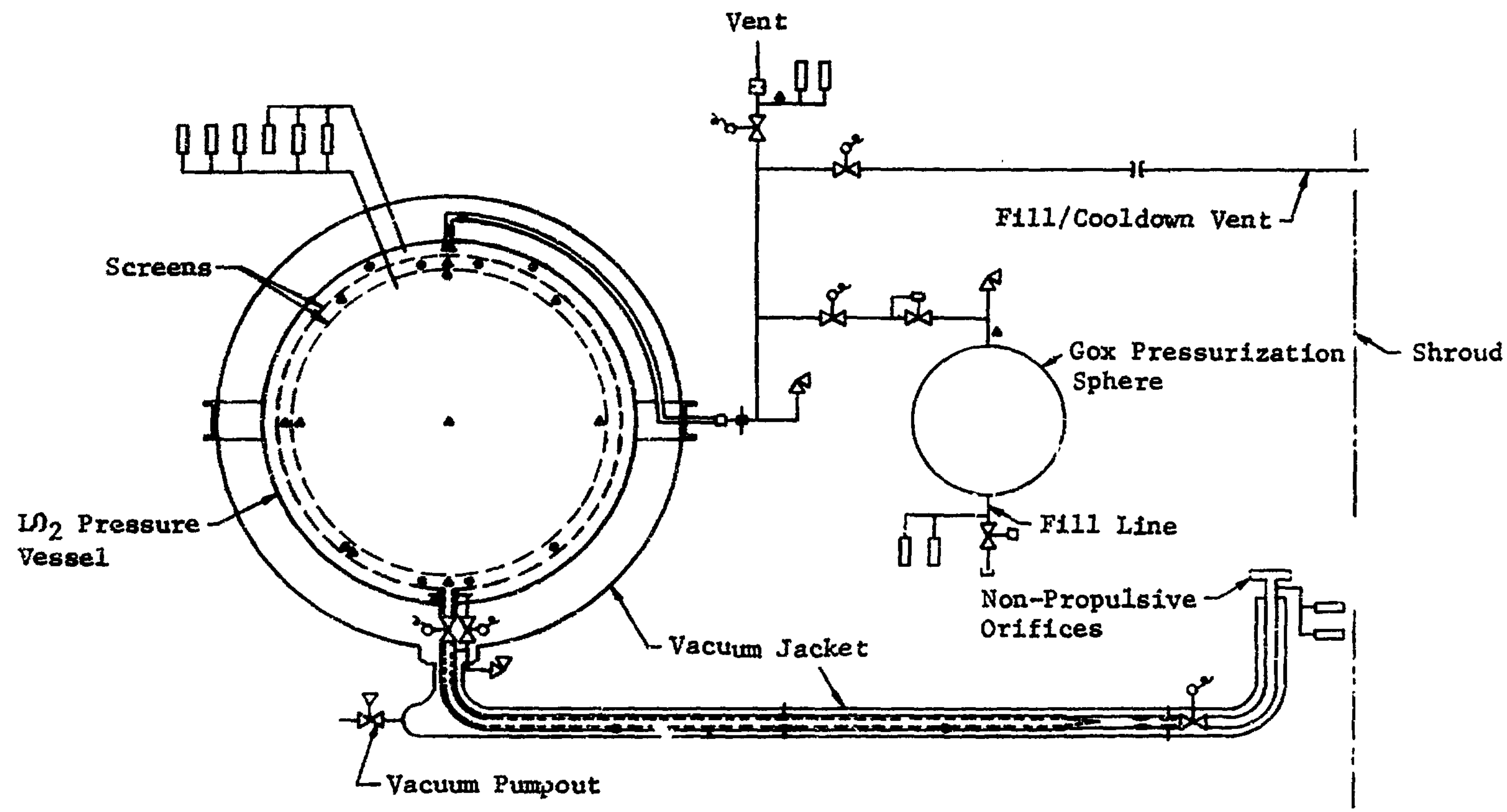
Filling is accomplished by evacuating the tank, breaking this vacuum with O_2 vapor, cooldown and filling with LO_2 saturated at near ambient pressure, and pressurizing with GO_2 to collapse vapor bubbles which may exist in the flow channels. Liquid level is determined by sensors located in the vapor annulus, flow channels, and bulk region; Figure III-28 shows these sensors.

c. *Vacuum Jacket* - High performance insulation is required for the LO_2 storage tank during the mission. Multilayer insulation (MLI) consisting of layers of mylar and nylon net under a high vacuum was selected. The vacuum jacket is fabricated from 6061 aluminum hemisphere spun to an ID of 91 cm (36 in.) from 0.48 cm (0.190 in.) plate. A 7.6 cm (3.0 in.) aluminum channel girth ring with a 2.5 cm (1 in.) web is sandwiched between the hemispheres. Support linkage for the propellant tank is attached to the girth ring. The girth ring is a support member for both the LO_2 storage tank inside and the entire assembly in the test module structure. The storage tank suspension linkage consists of three sets of three each 1.3 cm (9/32 in.) dia by 25 cm (10 in.) long stainless steel rods making a total of nine suspension points on the propellant tank and three equally spaced attachment points on the girth ring. The three rods of each set attach to the propellant tank at 120° spacing as shown in Figures III-29 and III-30.



Channels Before Screen Installation

Fig. III-27 Flow Channel Structure



Legend

- ● Liquid Sensor
- ▲ Platinum Temp. X-Ducer
- ⊣⊢ Capacitance Quality Meter
- Pressure X-Ducer
- ΔP X-Ducer

Fig. III-28 Test Article Instrumentation Schematic

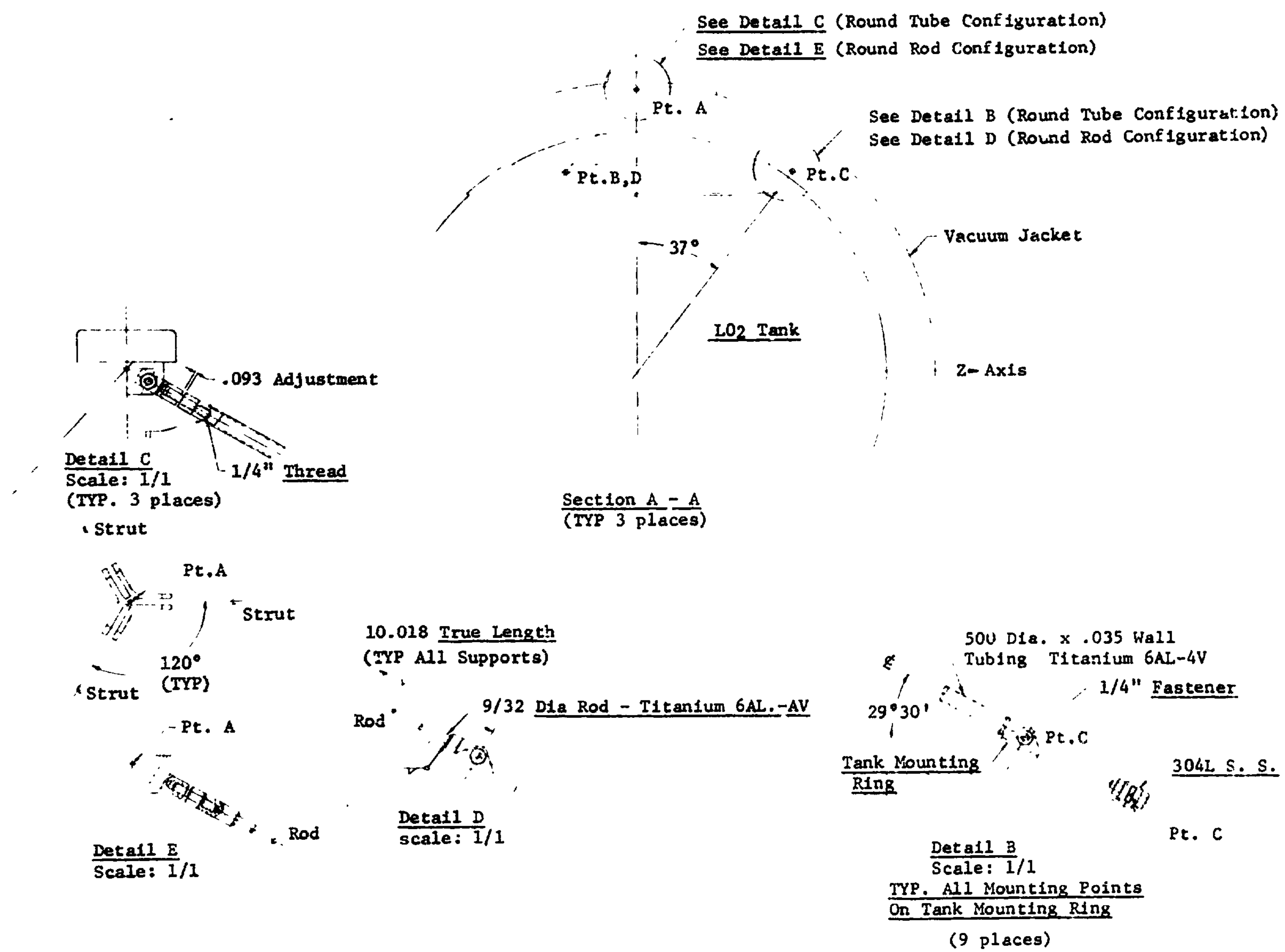


Fig. III-29 Storage Tank Suspension Detail

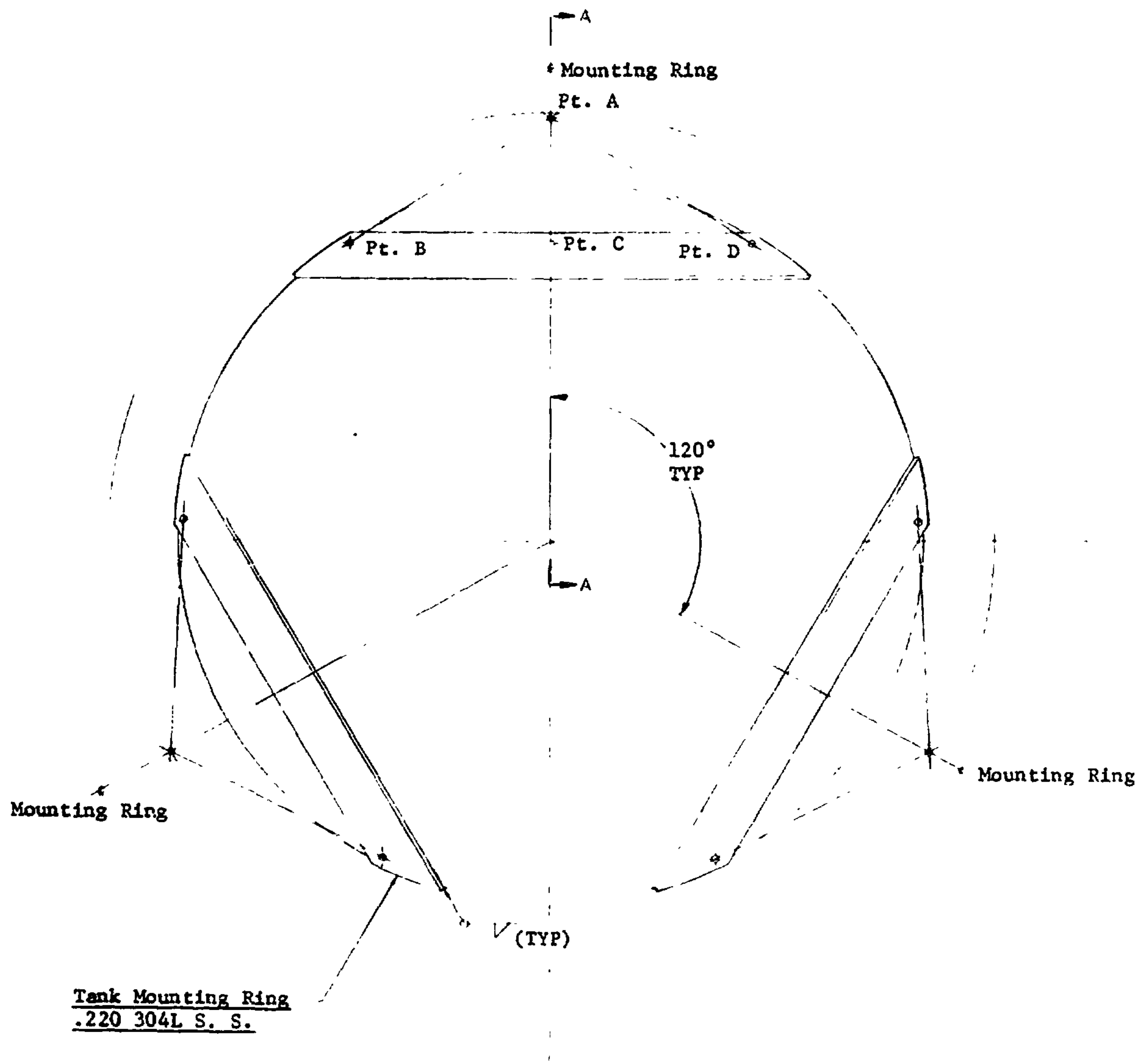


Fig. III-30 Storage Tank Suspension Geometry

The vacuum jacket has two penetrations. These are the vent/pressurization tube and feedline, which require stainless steel to aluminum transition sections. These transitions are commercially available, state-of-the-art items.

d. Feedline - The configuration selected for the feedline permits an evaluation of both a wet or dry line. In a wet line, a liquid core is maintained during coast and provides vapor-free liquid at the feedline outlet on demand. Figure III-16 shows the feedline containing a micromesh screen core in the line.

Two layers of 325 x 2300 mesh stainless steel screen are used in the feedline to increase the pressure retention capability. During liquid expulsion, viscous flow losses impose an additional pressure retention requirement on this screen to prevent vapor ingestion from the vapor annulus. The screen core terminates in a conical bubble strainer located upstream from the flow/quality meter. A non-propulsive discharge is achieved at the outlet by the flow-limiting orifices located in the tee section (Fig. III-28). A 10 cm (4.0 in.) vacuum jacket surrounds the 3.8 x 0.89 cm (1.5 in. x 0.035 in.) feedline wall which contains a 1.9 cm (3/4 in.) screen core. Capacitance-type quality meters are placed in the tank outlet, at the screen core midpoint, and upstream from the outflow valve. The presence of liquid or vapor in the vapor annulus is determined by liquid level sensors positioned as shown in Figure III-28.

e. Fill Line - The feedline is used for filling the LO_2 storage tank. Liquid oxygen, dewar storage-to-feedline connection is accomplished manually through an access door in the shroud. The tee section on the feedline is removed and the storage discharge line is adapted to the exposed B-Nut fitting.

f. Vent System - The vent system is designed to handle vapor venting from the LO_2 storage tank during both low-g orbital flight and prelaunch hold. The vent line penetrates the top of the storage tank and is routed through one quadrant of the vacuum annulus, penetrating the girth ring channel in the vacuum jacket. As shown in Figure III-23, the vent line size within the vacuum jacket is 2.5 cm (1.0 in.) dia and is reduced to 1.3 cm (1/2 in.) dia downstream. The 2.5 cm (1.0 in.) section carries the instrumentation leads for the temperature and liquid level sensors located in the storage tank. This approach is used to reduce the number of penetrations in the system and to reduce the heat leak conducted through the instrumentation leads. The leads are brought out of the vent line through Deutch connectors downstream from the girth ring penetration.

Venting during cooldown, fill, and pad hold is through the 1.3 cm (1/2 in.) dia line routed through the shroud access door and to launch pad vent system. A pull-away type disconnect is used for the on-pad vent. Venting during orbit flight is through a 0.10 cm (0.040 in.) dia orifice located downstream from the vent valve (Fig. III-23).

The vent pressure range should be maintained within the bubble point of the screen liner. If the vapor annulus pressure falls below the bulk region pressure, liquid oxygen will enter the vapor annulus. This condition should be avoided for venting efficiency. The vent control senses this pressure differential and activates the vent valve to maintain a pressure delta within the bubble point range. This pressure differential range for the 250 x 1370 mesh screen used on the liner is 2,680 N/m^2 (0.39 psi) for liquid oxygen.

The presence of liquid in the vent line is determined by three liquid level sensors located in the vent line near the storage tank and by a capacitance-type quality meter located in the line near the girth ring penetration. Two pressure transducers (one redundant) and one platinum temperature sensor are used to determine the vent flowrate through the 0.10 cm (0.040 in.) vent orifice (Fig. III-28).

g. *Pressurization System* - Gaseous oxygen was selected as the pressurant for the flight test article. The system is composed of storage sphere, fill line, regulator, relief valve, shutoff valve, and a 0.63 cm (1/4 in.) dia line fed into the vent line (Fig. III-28). Material compatibility with gaseous oxygen dictated the use of stainless steel for the storage sphere. Storage pressure is 1720 N/cm^2 (2500 psi) in a one-cubic foot sphere weighing 63 kg (140 lbm). The regulator, in series with the 0.63 cm (1/4 in.) electric solenoid shutoff valve, operates over a 5 to 1 pressure ratio which limits storage pressure to 345 N/cm^2 (500 psi) as a minimum to satisfy pressurant flowrate requirements. A preset downstream pressure of 28.0 N/cm^2 (40 psi) was selected for operating pressure. The 28 Vdc solenoid valve is controlled by ground command for pressurization prior to propellant expulsion. The storage container is filled before launch through the 0.95 cm (3/8 in.) dia line and hand valve. After filling, the line upstream of the valve is capped as an additional safety measure and as a backup for leaks from the shutoff hand valve.

Analysis shows a need for insulation of the storage sphere to maintain a desirable stored gas temperature ($\geq$ 400°R). High performance MLI, laminates of mylar and nylon net, were selected for this application. Instrumentation for the pressurization system consists of two pressure transducers (one redundant) with 0 - 21 x 10^6 N/m^2 0-3000 psi range and one platinum resistance type temperature sensor, 450K to 588K (350 to 600°F) as shown in Figure III-28.

2. Module Structural Design

The structural design of the test module was primarily influenced by the requirement to simulate the mass properties and dynamic response of the VLDS. These design requirements were presented in Section B. The flight test module structure consists of triangular members to provide the desired rigidity. As shown in Figure III 31, the basic truss contains triangles at the top and bottom formed from 7.6 x 7.6 x 0.63 cm (3.0 x 3.0 x 0.25 in.) wall square 6061 aluminum tubing. The triangles are rotated 60° with respect to each other about a longitudinal axis and are connected at the points of each triangle by square tubed members to form a rigid structure.

Some of the supporting subsystems, such as batteries and the GO_2 pressurant tank, are mounted on arms extending from each side of the lower triangle. This mounting procedure was used to obtain the required mass properties, c.g., and moments of inertia. The module must be confined to an envelope 380 cm (150 in.) in dia to provide adequate rattle space within the Centaur shroud. All GSE service connections are routed through the 30 cm x 38 cm (12 in. x 15 in.) access door located in the Centaur shroud.

3. Thermal Control Subsystem

a. Ground-Hold Thermal Control - An air conditioning system is provided within the Centaur shroud. As a result, the only significant thermal control concern is the test article itself. Venting of the LO_2 tank would be allowed during the pad hold time. At some point before liftoff, the vent line is closed and remains so until final orbit is achieved. After reaching the terminal orbit, venting will be activated, as required, to maintain tank pressure within a predetermined band and to satisfy test objectives.

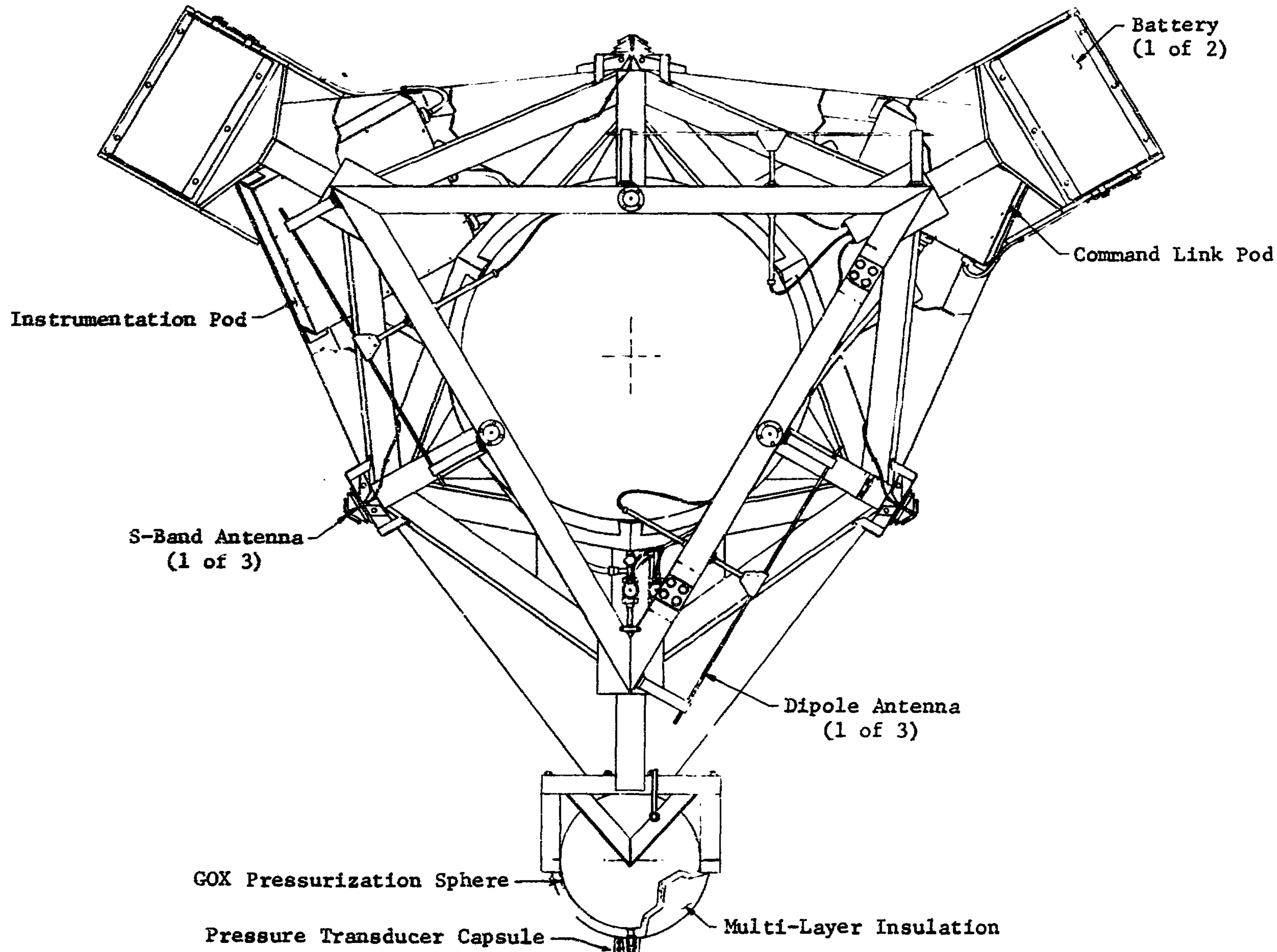


Fig. III-31 Flight Test Module-General Arrangement Top View

Undesirable slosh modes on ascent are precluded by limiting the ullage volume fraction in the test article to less than 5%. The unvented pressure rise during pad hold and ascent must be limited to less than 10.4 N/cm^2 (15 psi). With stratification of the LO_2, a larger pressure rise can be tolerated than if the liquid were well mixed, since this pressure will collapse when thermal equilibrium is approached.

To achieve the desired low heat-leak to the system, both the cryogen tank and feedline are vacuum jacketed. The heat paths between the vacuum jacket and cryogen are kept low. The tank is supported within the vacuum jacket by nine stainless steel rods. The feedline valves are completely enclosed within the vacuum jacket. The conduction heat transfer to the cryogen tank via the vent line is minimized by making the connections to the vacuum jacket and cryogen tank 90° apart, as seen in Figure III-23. The instrumentation leads are brought through the vent to allow vapor cooling during periods of venting.

Several insulations were considered and are acceptable in meeting the desired performance. The best performance is obtained with MLI which provides the greatest thermal isolation. Figure III-32 shows this performance. The best of the single, constituent powder insulations was the microsphere insulation produced by the Minnesota Mining and Manufacturing (3-M) Company. This powder insulation material consists of hollow glass spheres with typical sizes ranging from 15 to 150 μ in diameter (Ref III-10). Filling the vacuum jacket with this material is approximately equivalent to one half-inch of MLI. Perlite insulation, which has approximately twice the thermal conductivity of the microsphere, would also be an acceptable alternative. The application of perlite is probably more difficult than the microspheres since it does not flow as readily and tends to settle. A 50-50 mixture by weight of Cab-o-sil, a silica aerogel, and aluminum powder has a thermal conductivity of roughly one-half that of the microspheres (Ref III-11). This material tends to separate with low frequency vibration. Although this separation might not be a problem, the aluminum particles alone are quite conductive and in high concentrations would produce a greatly increased heat leak. Further, finely divided aluminum powder is quite flammable and perhaps should be avoided. For these reasons the Cab-o-sil and aluminum powder mixture was not considered. Low-density foam insulations are too conductive for this application and were not considered.

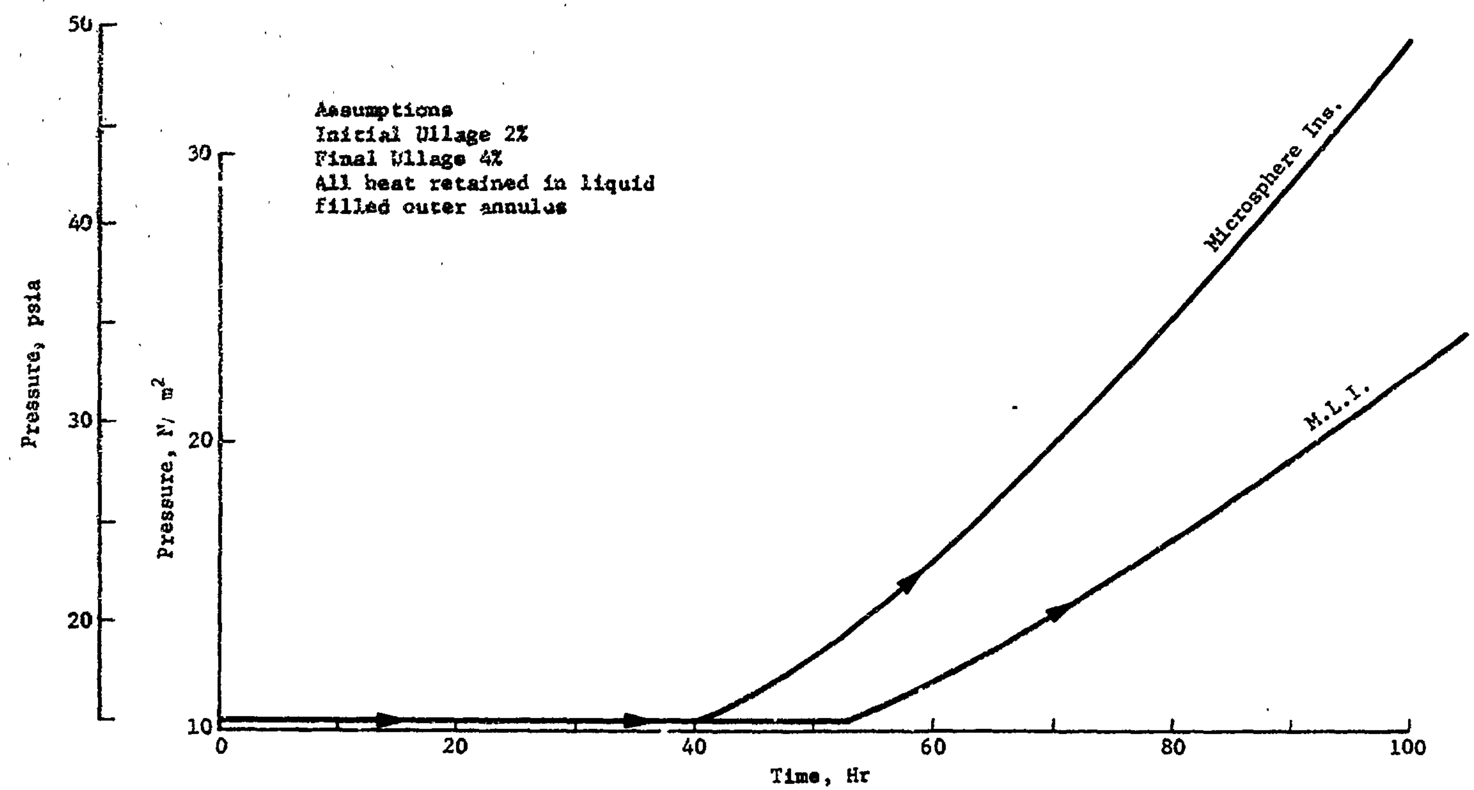


Fig. III-32 Vent Process Followed By Unvented Pressure Rise For Two Insulations

Based on the numbers shown in Table III-8, and the foregoing discussion, the clear choice was either the MLI or microspheres. The information, from Table III-8 and Figure III-22, is presented in a different format in Figure III-32 for these two insulations. Note that the time to vaporize 2% of the propellant is 40 and 53 hr for the microspheres and MLI, respectively. If the vent were then closed and the pressure allowed to rise from 10.4 to 21.0 N/cm^2 (15 to 30 psi), an additional 32 or 41 hr would elapse for the microspheres or multilayer, respectively. The longer times associated with the MLI lead to a greater flexibility in the ground hold operations. For this reason, the MLI was chosen for the baseline design.

b. *On-Orbit Thermal Control* - The thermal control subsystem is required to ensure an adequate thermal environment for the orbital experiment. External surface finish, thermal resistances built into the structure, and thermostat-heater systems were considered. In order to maximize the reliability and minimize development costs, the design was based as much as possible on passive elements using heaters as a backup. The use of active thermal control elements, such as louvers, was not considered.

Items requiring thermal control include: electronics, batteries, antennae, GO_2 pressurization sphere, and the LO_2 sphere. The electronics are contained in two rectangular boxes located within the module truss envelope. The batteries and pressurization sphere are located outside of the truss envelope, at the apexes of the triangle which appears when the vehicle is viewed along its longitudinal axis, Figure III-31. Three transmitting antennae, a slotted cone and turnstile configuration, are located around the periphery of the vehicle. The receiving antennae are dipoles. The test article is centrally located within the truss with liquid and gas overboard dumps located outboard of the truss envelope.

The thermal control design reduces energy losses to a low level and absorbs a relatively constant fraction of the incident solar energy, somewhat independent of vehicle orientation. The design approach is to shroud the orbital test module with a thermal blanket, and to use low thermal conductance attachments between the module structure, electronics, and batteries (see Fig. III-31). MLI is required around the batteries and the pressurization sphere. An external surface finish to be specified will result in a structure temperature below the set point of the heater thermostats located in the batteries and the electronics packages. Thus, there will be a constant low level flow of energy from the temperature-controlled packages to the structure. This leads to a conservative design since an error in the temperature prediction or an off-design condition will not necessarily preclude the ability to reject heat when the electronics are active.

The internal surfaces of the thermal blanket are painted black to enhance internal radiation. Good radiation exchange will promote uniform internal temperatures. The blanket will be composed of one to three layers of Schjeldahl laminate, X-850, which consists of two layers of aluminized mylar with a dacron net between. The weight of the laminate is 150 gr/m^2 (2.3 oz/yd^2). The shroud will be made in conveniently shaped segments and attached to the truss with Velcro fasteners to allow easy access to the interior.

The flow control orifices for both the gas vent and liquid outflow lines will be in good thermal contact with the structure. In this way, the structure will act as a heat sink and help maintain the orifice temperatures above the triple point.

A thermal resistance will be built into the structure interface between the experiment and the Centaur vehicle. This thermal resistance will allow the experiment to assume a temperature which is more independent of the Centaur with no large energy transfers.

4. Instrumentation

Platinum resistance sensors were selected for all temperature measurements on the module. These sensors require excitation by a voltage-regulated power supply and produce a full-scale output signal of approximately 20 millivolts. For liquid level sensing, a heated-resistance-element transducer type was chosen. An associated signal conditioning module requires a supply voltage of 18 Vdc and produces a bilevel type output signal of either zero or 18 volts, signifying either presence or absence of liquid at the transducer. Strain gage transducers with integral amplifiers were selected for pressure measurements. With an unregulated supply voltage of 28 Vdc, such a transducer produces a 0-5 volt analog output signal. Flowrate and fuel quality would be measured with a flowmeter containing a turbine to measure volumetric flowrate and a capacitance sensor to determine the density and quality of the liquid. The turbine and capacitance sensor are contained in the same housing. Integral signal conditioning is incorporated with a power requirement of 28 Vdc, and producing 0-5 Vdc analog output signals.

The basic design of the experiment made replacement of failed transducers impractical. Therefore, numerous redundant measurements were designed into the system so that a large number of failures could be tolerated.

Module housekeeping measurements consisted of temperatures, voltages, operating states, and battery current. Temperature measurements were specified to be made with the same type of sensor used in the experiment. For voltage measurements, voltage dividers were planned to reduce the signal range to the required multiplexer input range. Operating states were to be monitored by sensing the presence or absence of primary supply voltage at various components, using voltage dividers to provide signals compatible with the bilevel channel input requirements of the multiplexer. Battery current measurement was implemented by incorporation of two current shunts in parallel. The voltage drop across the shunts would produce a signal proportional to current; two shunts in parallel were used so that failure of one shunt would not disable the power system.

5. Data Acquisition Subsystem

A subsystem capable of acquiring and transmitting the measurements listed in Table III-14 was designed. A schematic diagram of the subsystem is shown in Figure III-33. This subsystem provided signal conditioning of outputs from transducers and other data sources, multiplexing of the conditioned signals, transmission of the multiplexed data, and command control of data acquisition sequences. Multiplexing was addressed first, because signal conditioning requirements are derived in part from the characteristics of the selected multiplexer.

a. Multiplexing - In the interest of minimizing cost, availability of surplus components from previous NASA programs was investigated. Preliminary inquiries established that the Orbiting Solar Observatory, Small Astronomy Satellite, and Biosatellite programs might be sources of applicable components. The first two programs were eliminated from consideration after it had been ascertained that only incomplete sets of hardware were available from these programs. Three nearly-complete sets of Biosatellite multiplexing and communications components were located at NASA Ames Research Center, and the decision was made to utilize this equipment.

The Biosatellite multiplexer consists of a commutator and a commutator-coder. Each unit samples about half of the total quantity of measurement channels, and the commutator-coder performs the additional function of encoding analog samples into binary words. Six-bit encoding is used, with an odd parity bit added to each 6-bit group. Total channel capacity consists of 114 analog channels sampled once per second, 16 analog channels sampled 8 times per second, and 36 bilevel channels sampled once per second. The encoding precision and sampling rates meet the requirements established for module measurements, and ample spare channel capacity is provided. The output signal to the telemetry transmitter is a 1792 bps biphase format pulse train.

Table III-14 Flight Measurement List

MEAS. NO.	DESCRIPTION	MEASUREMENT RANGE	SIGNAL RANGE
1.	Temp. Top Manifold	100-130 Deg K	0/20 MVDC
2.	Temp. Bottom Manifold	100-130 Deg K	0/20 MVDC
3.	Temp. Liquid Channel No 1	100-130 Deg K	0/20 MVDC
4.	Temp. Liquid Channel No. 2	100-130 Deg K	0/20 MVDC
5.	Temp. Liquid Channel No. 3	100-130 Deg K	0/20 MVDC
6.	Temp. Liquid Channel No. 4	100-130 Deg K	0/20 MVDC
7.	Temp. Communication Screen No. 1	100-270 Deg K	0/20 MVDC
8.	Temp. Communication Screen No. 2	100-130 Deg K	0/20 MVDC
9.	Temp. Communication Screen No. 3	100-270 Deg K	0/20 MVDC
10.	Temp. Pressurant Inlet No. 1	100-130 Deg K	0/20 MVDC
11.	Temp. Pressurant Inlet No. 2	100-270 Deg K	0/20 MVDC
12.	Temp. Vent Line Orifice	130-360 Deg K	0/20 MVDC
13.	Temp. Press. Regulator Inlet	130-360 Deg K	0/20 MVDC
14.	Liquid Level Top Manifold No. 1	On/Off	0/18 VDC
15.	Liquid Level Top Manifold No. 2	On/Off	0/18 VDC
16.	Liquid Level Bottom Manifold No. 1	On/Off	0/18 VDC
17.	Liquid Level Bottom Manifold No. 2	On/Off	0/18 VDC
18.	Liquid Level Channel Outlet No. 1	On/Off	0/18 VDC
19.	Liquid Level Channel Outlet No. 2	On/Off	0/18 VDC
20.	Liquid Level/Vent Outlet	On/Off	0/18 VDC
21.	Liquid Level Vapor Annulus Bottom	On/Off	0/18 VDC
22.	Liquid Level Feedline Ann. No. 1	On/Off	0/18 VDC
23.	Liquid Level Feedline Ann. No. 2	On/Off	0/18 VDC
24.	Position Valve H-3	Open/Closed	0/28 VDC
25.	Position Valve G-3	Open/Closed	0/28 VDC
26.	Position Valve L-1	Open/Closed	0/28 VDC
27.	Position Valve L-2	Open/Closed	0/28 VDC
28.	Press. Bulk Reg./Gas Ann. No. 1	±6900 N/Sq. M	0-5 VDC
29.	Press. Bulk Reg./Gas Ann. No. 2	±6900 N/Sq. M	0-5 VDC
30.	Press. Bulk Region No. 1	0-520,000 N/Sq. M	0-5 VDC

Table III-14 (c ncl)

MEAS. NO.	DESCRIPTION	MEASUREMENT RANGE	SIGNAL RANGE
31.	Press. Bulk Region No. 2	0-520,000 N/Sq. M	0-5 VDC
32.	Press. Pressurant Tank Fill Line No. 1	0-24 x 10^6 N/Sq. M	0-5 VDC
33.	Press. Pressurant Tank Fill Line No. 2	0-24 x 10^6 N/Sq. M	0-5 VDC
34.	Flowrate Liquid Line Downstream	0.07-0.7 Cu. M/Hr	0-5 VDC
35.	Flowrate Liquid Line Middle	0.07-0.7 Cu. M/Hr	0-5 VDC
36.	Flowrate Liquid Line Outlet	0.07-0.7 Cu. M/Hr	0-5 VDC
37.	Flowrate Vent Line	0.07-0.7 Cu. M/Hr	0-5 VDC
38.	Press. Upstream Vent Orifice No. 1	0-520,000 N/Sq. M	0-5 VDC
39.	Press. Upstream Vent Orifice No. 2	0-520,000 N/Sq. M	0-5 VDC
40.	Press. Upstream Outflow Orifice No. 1	0-520,000 N/Sq. M	0-5 VDC
41.	Press. Upstream Outflow Orifice No. 2	0-520,000 N/Sq. M	0-5 VDC
42.	Voltage Battery No. 1	0-36 VDC	0-36 VDC
43.	Voltage Battery No. 2	0-36 VDC	0-36 VDC
44.	Temp. Battery No. 1	250-350 Deg K	0-20 MVDC
45.	Temp. Battery No. 2	250-350 Deg K	0-20 MVDC
46.	Temp. Transmitter No. 1	250-350 Deg K	0-20 MVDC
47.	Temp. Transmitter No. 2	250-350 Deg K	0-20 MVDC
48.	Temp. Signal Conditioner No. 1	250-350 Deg K	0-20 MVDC
49.	Temp. Signal Conditioner No. 2	250-350 Deg K	0-20 MVDC
50.	Voltage 10 VDC Power Supply	0-11 VDC	0-11 VDC
51.	Battery No. 1 Heater Power	On/Off	0-28 VDC
52.	Battery No. 2 Heater Power	On/Off	0-28 VDC
53.	Equip. Compt. #1 Heater Power	On/Off	0-28 VDC
54.	Equip. Compt. #2 Heater Power	On/Off	0-28 VDC
55.	Current 28 VDC Bus	0-10 Amp	0-50 MVDC

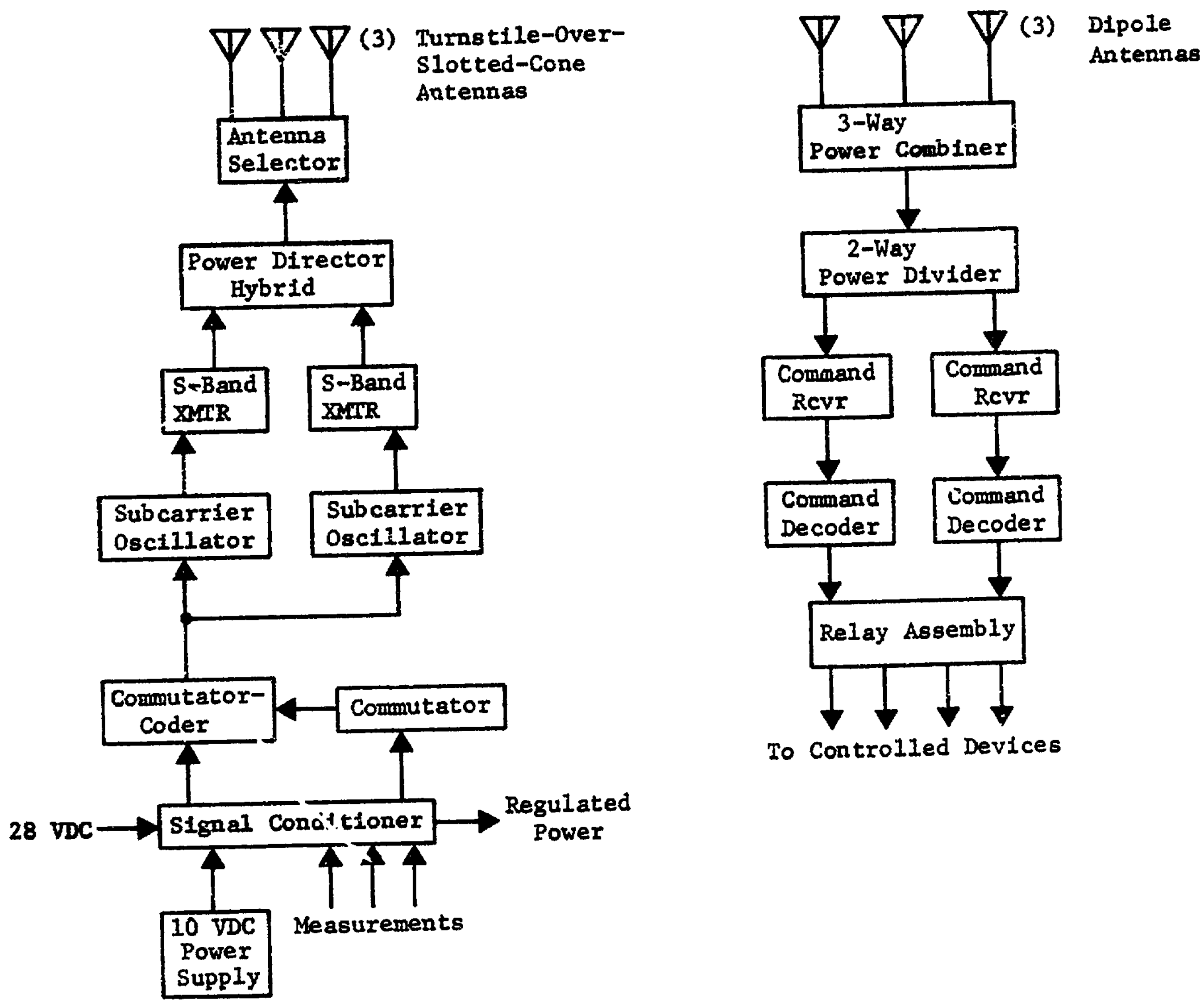


Fig. III-33 Data Acquisition Subsystem Block Diagram

Analog signals in the range 0-5 volts are required; bilevel signals between 0 and 2 volts are coded "0" and signals between 4 and 7 volts are coded "1". These parameters, together with the characteristics of the selected transducers, define the requirements for signal conditioning.

b. Signal Conditioning - Signal conditioner assemblies are nearly always designed to the unique requirements of a specific program, and a surplus unit suitable for this program was, therefore, not to be found. A decision was reached to design and fabricate the signal conditioner in-house, incorporating procured modular subassemblies wherever possible.

The approach to conditioning the various types of signals is shown in Figure III-34. The dc amplifiers and liquid level conditioners would be procured as potted modules, and would be integrated with the other components on printed circuit boards. A container was planned to provide secure mounting for the printed circuit boards, facilitate interconnections, and permit modular replacement of failed channels. Adequate spare channels were provided to allow for a moderate growth of measurement requirements. Controls shown in the signal conditioner circuitry provide adjustments of balance, span, and gain in temperature measurement channels.

Voltage regulator modules were also designed for incorporation in the signal conditioner assembly. These were required to provide regulated power to the commutator, commutator-coder, command receivers and decoders, and to signal conditioning modules. Potted modular regulators were identified which could be installed on the same size printed circuit boards used for signal conditioning channels. A Titan 10 Vdc power supply was selected to provide excitation voltage for temperature transducers.

c. Telemetry - As described in the Data Acquisition Analysis section, trade studies had established the use of S-band telemetry, with a transmitter output of at least 8.7 watts. To provide high reliability, two identical 10 watt transmitters were incorporated in the data acquisition system. Each transmitter was modulated by an Inter-Range Instrumentation Group (IRIG) band "E" subcarrier oscillator. The pulse code modulation (PCM) signal from the commutator-coder was fed to both subcarrier oscillators. Either transmitter could be activated for data transmission by commanding application of primary power to the selected unit.

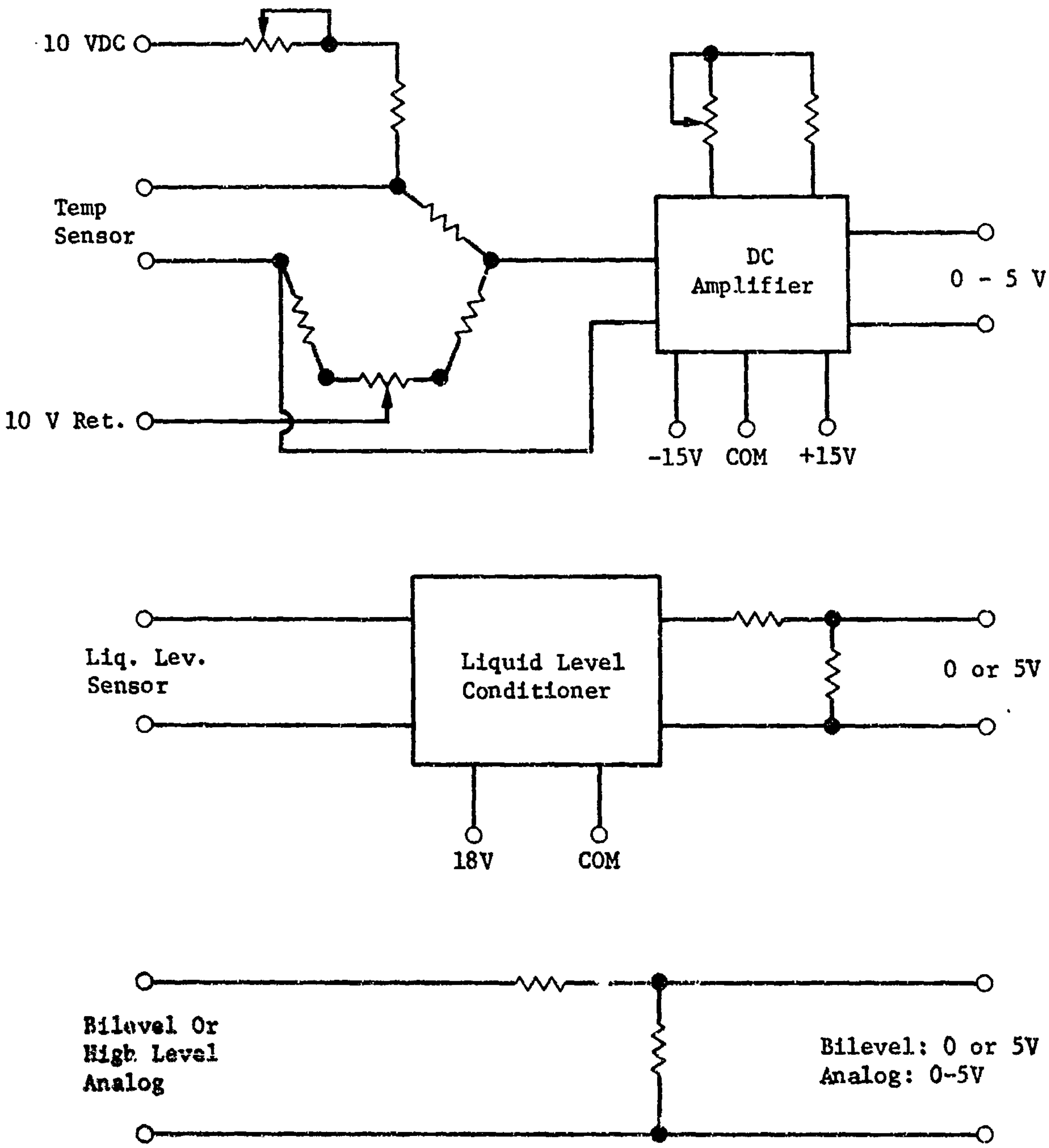


Fig. III-34 Signal Conditioning Circuitry

The telemetry antenna selected for the module is called a "turnstile-over-slotted-cone." This antenna was developed for use in the Viking Lander radar system at a frequency of approximately 1000 MHz. No problems were anticipated in scaling the design for use at 2250 MHz. The desirable feature of this antenna is its broad pattern; a gain of greater than unity is provided at angles within 75 degrees of the principal axis. Even with this broad pattern, it was deemed advisable to use 3 such antennas. This was done to enhance the probability that the randomly-oriented vehicle would have at least one antenna pattern intercepting the line-of-sight (LOS) to a ground station at any given time.

As a consequence of the broad pattern, interference peaks and nulls would be produced if two or more antennas were to be fed from the same transmitter. Accordingly, an antenna selector was designed into the system. This device would be operated through the command link to select the antenna producing the strongest signal at the ground station receiver. Solenoid-operated switches capable of handling 10 watts of power at S-band frequencies have been qualified for use in several space programs and are available.

A device identified as a power director hybrid was designed to route the output of either transmitter to the antenna selector. Completely passive, the power director hybrid imposes no more than 1 db loss on the signal from the active transmitter while maintaining an appropriate termination at the port to which the standby transmitter is connected.

d. Command - The trade study cited previously also selected a nominal frequency of 150 MHz for the command link. Biosatellite command receivers, operating at 148.98 MHz, were selected. Parallel redundancy, with two units active at all times, was specified to enhance reliability. As the system was designed, each receiver would provide a demodulated command signal to an associated command decoder (which is also a surplus Biosatellite component). Decoded commands would energize a relay assembly to apply or interrupt primary power to the controlled devices.

The tone digital command format for which the receivers and decoders were designed is a Spaceflight Tracking and Data Network (STDN) standard. Up to 70 different commands are feasible with this system. As shown in Table III-15, only 15 commands were anticipated to be required.

Table III-15 Orbital Test Commands

1.	Master Enable
2.	Valve H-3 Open
3.	Valve H-3 Close
4.	Valve G-3 Open
5.	Valve G-3 Close
6.	Valve L-1 Open
7.	Valve L-1 Close
8.	Valve L-3 Open
9.	Valve L-3 Close
10.	Data System On
11.	Data System Off
12.	Transmitter No. 1 On, No. 2 Off
13.	Transmitter No. 2 on, No. 1 Off
14.	Transmitters Off
15.	Antenna Selector Advance

An array of 3 dipole antennas, equally spaced around the roll axis, was chosen to provide a nearly-omnidirectional receiving pattern for the command link. A 3-way power combiner and a 2-way power divider were specified to sum the outputs of the 3 antennas and to distribute half the summed signal to each command receiver.

6. Power Subsystem

Analysis of the module electrical power profile yielded an energy requirement of approximately 220 amp-hr for the nominal one-week mission. Reserve capacity sufficient to extend the battery life to two weeks was identified as a desirable design objective. A timeline of activities for a second week in orbit was not developed, but there was general agreement that the energy requirement would be considerably lower than that for the first week. For battery sizing purposes, a two-week energy requirement of 330 amp-hr was assumed.

Since size and weight were not constraining, two batteries with diode isolation could be considered. It would be mandatory that either battery could supply the one-week energy requirement in the event the other battery failed. Capability to complete a two-week mission on one battery would be desirable but not mandatory.

A survey of space-qualified batteries available as surplus from previous programs revealed the existence of a number of batteries built for use in the Apollo Lunar Module (LM) descent stage. The LM battery, manufactured by Eagle-Picher Industries, Inc., is a manually-activated, silver-zinc unit with a nominal capacity of 400 amp-hr. It is identified by Eagle-Pitcher part number MAP 4324-013. Although the energy capacity of this battery is considerably in excess of the module requirements, the overriding consideration of minimizing cost led to its selection.

However, the high energy capacity presented a design problem. Immediately after activation, with only the command receivers energized, a battery voltage of 37 volts was predicted. To avoid exceeding the power dissipation limits of module components, it would be necessary to keep the power bus voltage below 32 volts.

Several candidate techniques for reduction of the battery voltage were considered. Pre-discharging part of the battery capacity after activation would reduce the terminal voltage, and could be considered because much of the capacity was surplus to mission needs. More than one isolation diode could be placed in series with each battery, to reduce the bus voltage approximately 0.5 volt per diode. This technique would have the disadvantage of effectively raising the endpoint voltage of the battery by the same amount, and thus would reduce the usable energy capacity. A zener diode bus regulator could be incorporated to limit the maximum bus voltage, but would dissipate a significant amount of power. A combination of all three methods was recommended, and a test program to optimize the design was planned.

The final design thus incorporated two-400 amp-hr batteries in a parallel redundant circuit to provide energy capacity of approximately twice the estimated requirement.

7. Preliminary Design Review (PDR)

A PDR was held at the Martin Company on December 19, 1972. As a result of this review, several changes and action items which would impact the design were established. Work on this phase of the contract was terminated December 27, 1972. Consequently, the action items were not completed. These items are listed below:

1) Obtain latest proof flight prelaunch timeline of events. Extend particular attention from T - 72 hr to T - 0 hr.

2) Establish best timeline for cryogenic orbital test and determine how it can best merge with Titan/Centaur timeline on a minimum interference basis.

3) Establish environmental criteria for flight test module payload, i.e., longitudinal "g," lateral "g," vibration, acoustics, and temperature under Centaur fairing.

4) Determine atmospheric "g" loads on test module at time of Centaur fairing jettison. This information is needed for thermal control curtain design.

5) Determine number of layers of MLI to be used on LO_2 tank and feedline to yield desired total heat leak.

6) Determine if raising LO_2 storage tank relief valve from 34.5 to 48.3 N/cm^2 (50 psia to 70 psia) increases tank mass to a level requiring repeating the dynamics analysis.

7) Review alternative methods of determining preloaded tension stress in LO_2 tank support rods.

8) Determine effect of welding vacuum jacket hemispheres to girth ring. Will distortion have adverse effect on support rod tension?

9) Select method of locking tightening nuts of LO_2 tank support rods.

10) Select method of measuring tension loads in LO_2 tank support rods.

11) Provide lugs on the girth ring to facilitate assembly, handling, and support equipment (ASHE).

12) Complete analysis of feedline valve opening and closing characteristics required to prevent feedline screen breakdown.

13) Review method of supporting dodecasphere capillary screen device. Brackets from each gore to LO_2 storage tank wall may permit too much freedom for motion.

14) Resolve selection of device, or combination of devices, to be used in monitoring feedline flow rate and quality. Document information available on each device.

15) Resolve transducer type, mechanical, or electronic, to be used in monitoring bulk liquid region/vapor annulus pressure differential. Perform bench tests to validate decision.

16) Determine insulation requirements for GO_2 pressurization sphere.

17) Determine adaptability of Simmonds capacitance device to monitor quality of flow in ventline while venting.

18) Determine minimum land lines to blockhouse necessary to satisfy LC-41 launch operations safety requirements.

F. INTEGRATED TEST PLAN

This plan describes the development and qualification test program for the orbital test module. The test program includes component, subsystem, and system tests for flight qualification and acceptance of hardware. Testing was to be performed with flight hardware, since no development hardware was to be built. The test flow chart in Figure III-35 shows the sequence of tests to be performed at both Martin Marietta and KSC. Component development and acceptance tests are shown in Table III-16a. Subsystem tests are summarized in Table III-16b.

1. Test Requirements

a. Development Tests - Development tests shall be performed where necessary to verify: (1) feasibility of the design approach; (2) performance capability of the subsystem; and (3) to acquire data to support the design and development process.

b. Flight Certification - Flight certification shall verify that flight hardware meets the performance and design requirements under the anticipated operational environments, plus allowable margins. Flight certification of all components shall be performed by test, analysis, or similarity to previous usage.

Components and assemblies will be flight certified at the highest assembly level identified as a remove and replace item from the flight vehicle. Flight certification levels (environmental and functional) shall be sufficiently higher than flight acceptance test levels to demonstrate that all components will perform within specification.

c. Flight Acceptance - Flight acceptance test shall consist of functional tests and environmental tests of components and assemblies to assure compliance with performance specifications and to demonstrate functional adequacy of component/assembly for flight use. Acceptance testing shall also include the test requirements for the assembled subsystems as well as the integrated system as necessary to verify and demonstrate their compliance with design specifications and operational integrity. As a minimum, flight acceptance tests will include:

1) visual and other non-destructive inspection for conformance to specifications, drawings, and workmanship standards;

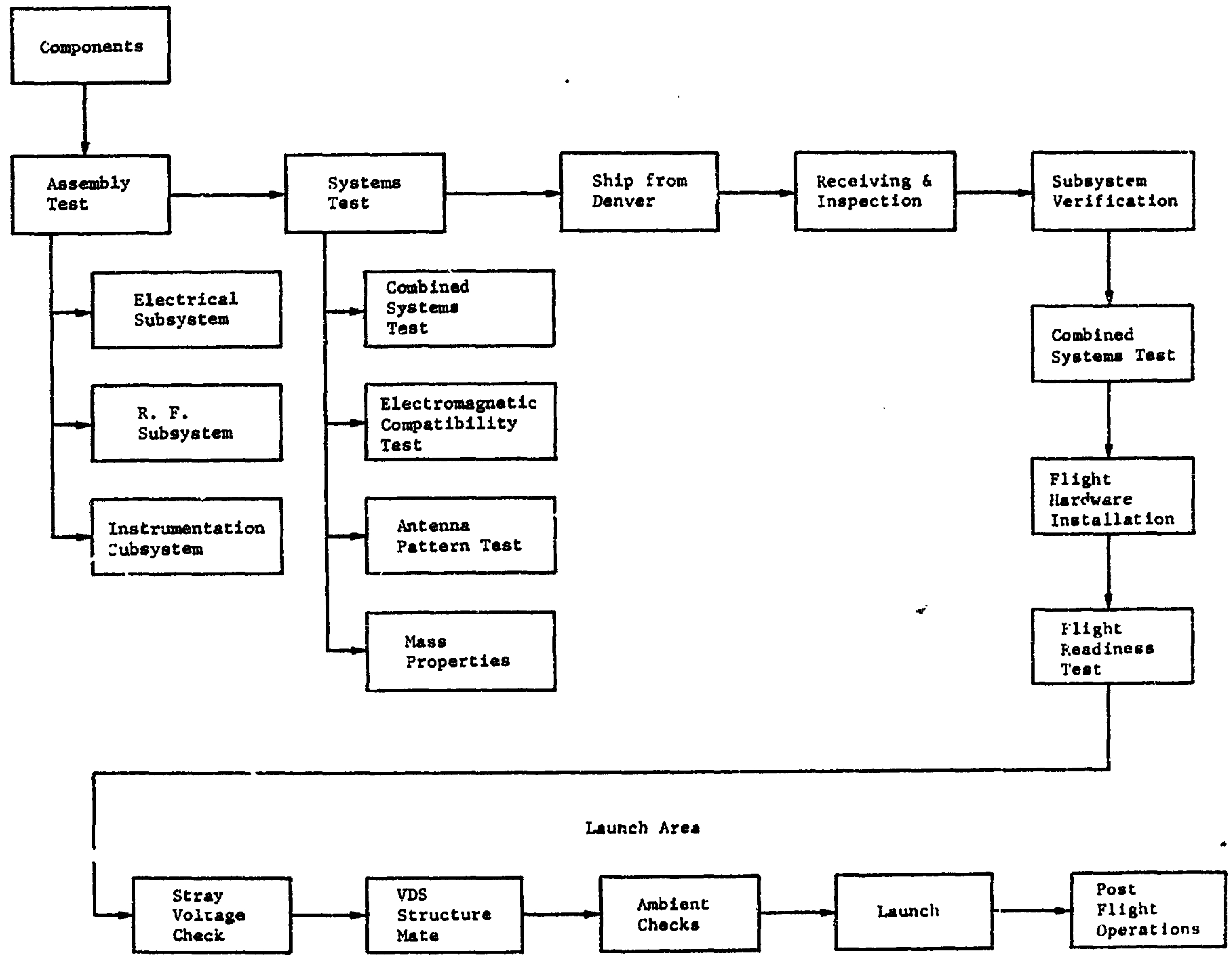


Fig. III-35 Experiment Module Test Flow

Table III-10a Component Development and Acceptance Tests

A = Analysis
S = Support
X = Test

						MMC DEVELOPMENT TESTS				ACCEPTANCE TESTS			
ITEM	MAKE OR BUY	GVS AT GDCA	SCALE MODEL	PATTERN TESTS	VIBRATION	TEMPERATURE	PROOF LOAD	FUNCTIONAL	INSPECTION	PROOF PRESSURE	TEMPERATURE CYCLING	FUNCTIONAL	VIBRATION
Press & Vent Subsystem									X				
Sphere Subassembly	M								X	X			
Sphere	B								X	X			
Cap	B								X				
Solenoid Valve	B								X			X	
Burst Diaphragm	B								X				
Pressure Regulator	B								X			X	X
Orifice	M								X				
Hand Valve	B								X			X	
Relief Valve	B								X			X	
Welded Lines	M								X				
Disconnect	B								X			X	
Heaters	B								X			X	
Propellant Subsystem									X				
Tank Assembly	M								X	X			
Hand Valve	B								X			X	
Flow Meter	B					X			X			X	
Relief Valve	B								X			X	
Pumpout & Relief Valve	B								X				
Tank Screens	B								X				
Tank Insulation	B								X				
Screen Tube	B								X				
Temperature Sensor	B								X			X	
Pressure Sensor	B								X			X	
Liquid Vapor Sensor	B								X			X	
Disconnect	B								X			X	
Solenoid Valve	B								X			X	
Truss Assembly	M						X		X				
GVS Test Article	M	S							X				
Payload Module	M								X			X	
RF Subsystem									X				
Command Antenna	M		X	X	X	X			X			X	
Telemetry Antenna	M		X	X	X	X			X			X	
Multiplexer	B								X				
Antenna Selector	B								X				
Power Divider	B								X				
Triplexer	B								X				
COAX	B								X			X	
Test Letter	A	B	C	D	E	F	G	H	I	J	K	L	M

Table III-16a (concl)

Test Letter	A	B	C	D	E	F	G	H	I	J	K	L	M
Power Subsystem Assembly	M								X				
Battery	B								X				
Heaters	B								X			X	
Power Subsystem Housing	M								X				
Cabling	M								X				
Communications Subsystem Assy.									X		X	X	
Telemetry Subsystem													
Signal Cond. Assembly	M				X	X		X	X		X	X	
Temperature													
Liquid													
Volt Divider													
DC Amp													
Commutator	GFP								X				
Commutator Coder	GFP								X				
S-Band XMTR	B								X		X	X	
10-7DC Pwr Supply (Titan)	B								X		X	X	
Command Subsystem													
Relay Assembly	M				X			X	X				X
Decoder	GFP								X				
Command Receiver	GFP								X				
Heaters	B								X			X	
Housing	M								X				
Cabling	M								X				
Electrical Test & Checkout Set	M								X			X	
Propellant & Pressurization													
Loading Set	M								X			X	
Handling & Support Equipment	M								X				

Table III-16b Denver and KSC Testing

A = Analysis
S = Support
X = Test

	MMC FLIGHT PROOF TESTS										KSC PRELAUNC TESTS		FLIGHT TESTS
ITEM	SLOSH	VIBRATION	THERMAL VACUUM	MICROMETEOROID	PRELAUNCH ENVIRONMENT	FUNCTIONAL	TEMPERATURE	SHOCK	EMI	SYSTEM FUNCTIONAL	T/M CHECKOUT	READINESS REVIEW	POST FLIGHT DATA ANALYSIS
Press & Vent Subsystem													
Sphere Subassembly													
Sphere													
Cap													
Solenoid Valve													
Burst Diaphragm													
Pressure Regulator													
Orifice													
Band Valve													
Relief Valve													
Welded Lines													
Disconnect													
Heaters													
Propellant Subsystem													
Tank Assembly	A												
Hand Valve													
Flow Meter													
Relief Valve													
Pumpout & Relief Valve													
Tank Screens													
Tank Insulation													
Screen Tube													
Temperature Sensor													
Pressure Sensor													
Liquid Vapor Sensor													
Disconnect													
Solenoid Valve													
Truss Assembly													
GVS Test Article													
Payload Module		A	X	A	A	X		A	X	X	X	S	S
RF Subsystem													
Command Antenna													
Telemetry Antenna													
Multicoupler													
Antenna Selector													
Power Divider													
Triplexer													
COAX													
Test Letter	N	O	P	Q	R	S	T	U	V	W	X	Y	Z

Table III-10b (cont'd)

Test Letter	N	O	P	Q	R	S	T	U	V	W	X	Y	Z
Power Subsystem Assembly													
Battery													
Heaters													
Power Subsystem Housing													
Cabling													
Communications Subsystem Assy.		X				X	X						
Telemetry Subsystem													
Signal Cond. Assembly		X				X	X						
Temperature													
Liquid													
Volt Divider													
DC Amp													
Communicator													
Communicator Coder													
S-Band													
10-VDC Pwr Supply (Titan)													
Command Subsystem													
Relay Assembly		X				X		X					
Decoder													
Command Receiver													
Heaters													
Housing													
Cabling													
Electrical Test & Checkout Set													
Propellant & Pressurization Loading Set													
Handling and Support Equipment													

2) exposure to mission related environments as necessary to reveal any component and/or assembly defects which might affect performance during the mission;

3) functional verification;

4) integrated systems tests to verify system interface requirements which cannot be verified at lower levels. System interface test requirements may be satisfied by means of planned use of simulators when mating item is not available;

5) verification of redundancy or alternate operating modes.

d. Test Policies - The following test policies are based on accepted procedures of Martin Marietta, modified as necessary to meet the specific constraints relating to acceptance testing of the orbital module.

1) The configuration of the TSE, ancillary equipment, and test procedures shall be verified and approved.

2) A pretest briefing shall be conducted to ensure that all test prerequisites have been satisfied.

3) All testing shall be performed in accordance with approved procedures. All test documents shall be prepared in accordance with the test criteria as specified in the System Test Specification.

4) System level functional tests shall be standardized to the extent practicable to provide continuity and to permit correlation of test results between test phases.

5) All nominal operating modes, included selected alternate modes, shall be demonstrated during flight acceptance testing.

6) A time log will be maintained throughout the checkout phase of the system against equipment time and/or cycle limit violations.

7) When a failure occurs and the problem is such that a hazardous condition to personnel or damage risk to the module and associated equipment is created, the test shall be stopped and the system including the TSE placed in a safe mode.

8) If any failure or malfunction of the orbital experiment module occurs, continuation of testing shall be determined by an investigation of the nature and cause of the failure or malfunction. The need for corrective action shall be determined and consequent retest requirements established prior to resumption of testing.

e. Data Requirements - The response of the module to the test command sequence and other test requirements during system level testing shall be monitored by the TSE. The following ground rules shall be used in establishing data analysis procedures.

1) Engineering test data shall be evaluated to assure that test parameters satisfy the test and checkout requirements.

2) Variations from procedural requirements will be recorded.

2. Component Tests

The tests presented in Table III-16b shall be performed to assure that all module components comply with the design and performance requirements of the experiment module.

a. Performance Requirements - Functional testing shall be performed during the component test program, as necessary, either to simulate the operative state of the equipment during an environmental exposure or to evaluate the effects on equipment performance subsequent to an environmental exposure. Functional tests prior to and following any environmental exposure shall require complete evaluation of all appropriate performance characteristics.

b. Flight Acceptance Test - Module component flight acceptance tests shall consist of functional tests of component and/or assembly in simulated flight level environment.

1) Temperature Cycling - the component and/or assembly, while in operating and non-operating modes, shall be exposed to the required temperature extreme for one hour minimum or as required to achieve temperature stabilization for a total of 1 1/2 cycles (low-high-low, three hours minimum).

2) Vibration - The component and/or assembly, while in operating or non-operating modes, shall be exposed to the indicated random vibration level for 30 seconds in each of the three orthogonal axes.

c. Flight Certification Test - All components shall be flight certified to the following minimum environmental conditions to verify that components and/or assemblies indicated in Table III-16a meet the expected flight environmental requirements. Flight certification shall be performed by test, analysis, as well as similarity to prior usage.

1) *Temperature Extremes* - The temperature extremes (non-operating and operating) are based on expected temperature environment for component locations. The temperature requirements will be satisfied either by component test to the temperature extremes or by appropriate thermal control.

2) *Pyrotechnic Shock* - Pyrotechnic shock certification of components shall be performed by a system level pyrotechnic shock test. Successful completion of the system level pyrotechnic shock test shall constitute satisfaction of component pyro-shock certification requirements of Table III-16a.

3) *Random Vibration* - The random vibration environment shown in Figure III-36 shall be applied to the specimen in each of the three major orthogonal axes for 6.7 sec per axis.

4) *Sustained Acceleration* - The test specimen shall be subjected to sustained acceleration at TBD g for one minute or as long as necessary to perform functional verification, in each direction of the three major orthogonal axes (six directions). Acceleration tests shall be performed only in th e necessary axes and direction as ascertained by component orientation within the vehicle.

5) *Seal* - The quality of the seal of hermetically-sealed or pressure-sealed components shall be verified prior to and after the component has been exposed to the environmental tests as required in Table III-16b. Two general test categories are acceptable for use: (1) High sensitivity and (2) Low sensitivity leak detection. High sensitivity may use any of four methods: (1) Vacuum Testing; (2) Pressure Testing; (3) Pressure-vacuum testing; and (4) Radiflo. Low sensitivity testing method shall be liquid immersion test.

6) *Flight Spares* - Components that have physically and successfully passed all of the flight certification environmental tests and the post-certification functional tests shall be considered as flight spares, and will not require flight acceptance testing. Those components that physically and successfully pass partial flight certification tests, shall also be considered as flight spares, but will require that the specified flight acceptance tests be performed. Flight acceptance tests will be performed after completion of the required certification tests. Those components which require pyro shock testing, at the component level, will require as a minimum, that a random vibration test also be performed. The random vibration test will be performed at flight acceptance levels, and will be performed subsequent to the pyro shock test, in order for these components to be considered as flight spares.

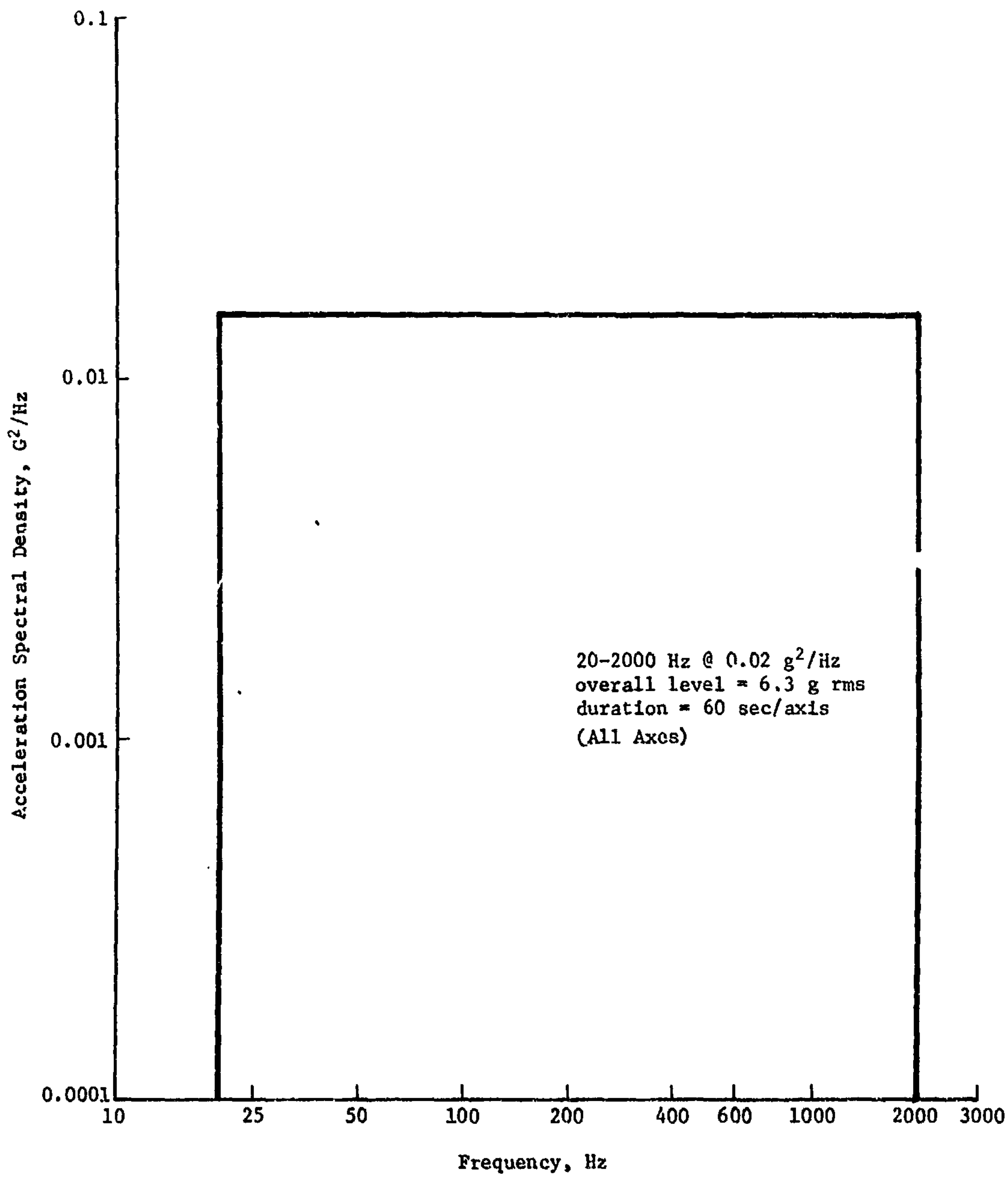


Fig. III-36 Random Vibration Acceptance Spectrum

3. Subsystem and System Tests

Test module acceptance testing at the Martin Marietta Denver facility will be performed at the highest subsystem level and at the integrated system level. Each subsystem will be checked out and functionally verified prior to integrating the subsystems into a combined system. System testing will be performed to verify subsystems interaction compatibility and provide total system functional verification.

a. *RF System Impedance Development Test* - The purpose of this test is to obtain antenna system parametric data which will be used to confirm the RF subsystem design or to provide data to determine a more effective arrangement of RF components. The RF impedance test will be performed using a scale size mock-up model of the module which contains the RF subsystem. The mockup model will be situated on the MMC antenna test range for the following tests:

1) radiation pattern measurements on transmitting and receiving antennas;

2) interference pattern versus antenna location;

3) RF interconnect cable configuration and phasing effects;

4) impedance and insertion loss measurements;

5) establish component location for optimum functional performance and compatibility.

b. *Electrical Subsystem Tests*

1) *Wiring Verification* - These tests will ensure the integrity of the airborne electrical system by verifying that the wiring and cordage have been manufactured to the engineering requirements. All wiring will be continuity and megger tested after installation. Cable and wire bundle requirements and cordage routing will be verified to conform with engineering drawings.

2) *Ground Verification* - This test will verify that the isolated electrical ground-return system is grounded at a single point only. With all module systems connected except batteries, the motor driven switches open, and the vehicle electrically disconnected from ground cabling, resistance between vehicle structure and the negative bus will be measured with the main structure ground point lifted. Resistance reading shall not be less than 10K ohms. With the main structure ground point connected, reading between negative bus, to structure shall not exceed 0.2 ohms.

3) *Power Isolation* - This test will verify proper ground isolation on the main power circuits prior to TSE connection and power application to the vehicle. With the batteries disconnected, isolation (10,000 ohms) shall be verified between battery output terminals and structure ground.

4) *Power Distribution and Load Verification* - This objective of this test is to verify the power transfer function and load distribution of the module electrical subsystem. With the module completely assembled, individual components or subsystems shall be sequentially turned on. Current measurements shall be recorded to obtain load characteristics. Power transfer function will be verified.

c. *Instrumentation Subsystem*

1) *Instrumentation Setup* - This test will verify instrumentation subsystem, component, and end instrument calibration. End instruments and transducers will be calibrated prior to installation. Commutator encoder will be calibrated by application of known inputs.

2) *Instrumentation Verification* - This test will verify the end-to-end functional integrity of the instrumentation subsystem. After the instrumentation equipment setup tasks have been completed, end instruments and transducers will be other stimulated or simulated with known input to the transducers and the TM transmitter output monitored via transmission line to the TSE rack or telemetered open loop to the TSE S-Band receiving antenna. Data commutation format and data decommutation verification will be made.

3) *Telemetry Transmitter Deviation Verification* - This test will verify TM transmitter deviation. TM transmission to the TSE rack will be made to verify that transmitter frequency deviation within performance requirements.

4) *Telemetry Transmitter Verification* - This test will verify TM transmitter power output and frequency. With the TM transmitter operating, power and center frequency measurements with no modulation will be made to verify performance requirements.

d. *Command Subsystem* - This test will verify the operational characteristics of the command subsystem. The experiment module command subsystem shall be connected closed loop to the TSE and commands transmitted. Verification shall include correct subsystem operation. Testing shall verify command receiver sensitivity.

e. TSE, and AHSE Verification

1) *TSE* - All electrical test support equipment (TSE), and associated cabling shall be functionally verified and/or calibrated prior to being electrically mated with the experiment module. As a minimum requirement, the test support equipment verification shall include the following tests: TSE rack mounted equipment checkout; TSE power supply voltage verification; interconnection cable set megger and continuity check; and RF transmission line verification.

2) *AHSE* - All assembly, handling, and support equipment (AHSE) will be load tested prior to use. Special AHSE will be tested to the requirements specified in the AHSE engineering drawings.

4. System Tests

a. Combined System Test (CST) - This test will verify experiment module system performance compliance with electrical and electromechanical design specifications. The CST will demonstrate that the integrated module system can, in the proper sequence, perform all simulated flight functions. The operational compatibility between the vehicle system and TSE will be demonstrated during the CST.

Upon completion of all subsystem tests, the subsystems will be integrated into a single system and a CST performed. It shall be verified that all nonpyrotechnic components and equipment perform their functions properly. The CST test sequence and format shall be designed to sequence through all commands and the corresponding response events will be monitored for a normal flight. Functional verification of the system redundancies will be performed.

A standard format for the CST will be developed such that the test and checkout criteria will satisfy test requirements for CST testing at Denver. Performing nearly identical CSTs will provide a data baseline which will permit verification of functional repeatability and system stability.

b. Pyrotechnic Shock Test - This test will evaluate the effects of the SPHINX pyrotechnic device activation on the module system performance. The experiment dummy simulator which is configured to simulate flight configuration shall be used for this test.

c. Antenna Pattern Test - This test will measure and record antenna system radiation patterns. The orbital module will be mounted on the antenna test fixture at the MMC antenna range where spherical contour antenna pattern measurements will be made.

d. Mass Properties - Upon successful completion of system testing, the vehicle shall be assembled to as near flight configuration for flight weight determination and establishment of longitudinal c.g. location.

5. Prelaunch Tests

Upon completion of the mass properties tests in Denver, the vehicle will be shipped to KSC for prelaunch and launch operations. The test requirements for the prelaunch and launch operations are outlined in Figure III-35. The TSE will be checked and verified prior to initiation of subsystem and system tests.

a. Subsystem Verification - Subsystem verification shall be performed to verify that the subsystems have retained their operational integrity as initially established and demonstrated during Denver testing. Upon completion of the receiving and inspection operation, a series of subsystem verification tests shall be performed to verify that the subsystems are operationally compatible to support combined system level tests.

The instrumentation subsystem verification shall be accomplished by performing the instrumentation setup and the end-to-end functional test. The RF subsystem verification shall be accomplished by performing the TM transmitter deviation test and TM transmitter verification. The command receiver and decoder shall be functionally verified by performing the command system checkout.

b. Combined Systems Test - The CST will verify that the module system can, in the proper sequence, perform all flight functions. The operational compatibility between the vehicle system and TSE shall be verified.

With the subsystems integrated into a total system a CST will be performed. Airborne battery power source will be simulated by ground power. The CST test format will sequence through all commands and the corresponding response events shall be monitored.

c. Flight Readiness Test (FRT) - This test will involve a final preflight functional verification of the module vehicle system. With the vehicle assembled in flight configuration, a CST will be performed. Flight batteries will be used for conducting the flight readiness test. The CST will sequence through all commands and their response will be monitored. The use of the airborne batteries shall be limited to five FRT runs or 10 minutes load time, whichever occurs sooner.

IV. GROUND VIBRATIONAL SURVEY MODEL

The opportunity for flying the capillary propellant management experiment on the Titan IIIE/Centaur proof flight was not pursued until mid-August 1972. As discussed in Chapter II, the dedicated launch scheme using an Atlas-F was the preferred orbital test method at contract initiation on August 9, 1971. The proof flight had been planned since early 1972 (Ref IV-1) with schedule milestones to launch the vehicle in January 1974. Incorporation of the orbital experiment into this plan was based, therefore, on causing no impact to this schedule. Discussions with the integrating contractor, General Dynamics Corp., Convair Astronautics Division (GDCA), in August 1972 (Ref IV-2), identified the Ground Vibrational Survey (GVS) test series of the VDS payload to begin in January 1973. After this test series, which was to last approximately five months, the payload was to be shipped to KSC for further preflight events prior to launch. The Viking Dynamic Simulator (VDS) as shown in Figure IV-1, consisted of a mass and dynamic simulator of the Viking spacecraft. The purpose of the payload, which was a secondary proof flight objective, was to obtain vibrational flight data for the Viking Lander Capsule Adapter (VLCA). The VLCA is a truss system that supports the Viking Lander Dynamic Simulator (VLDS) on the Viking Orbiter Dynamic Simulator (VODS). As proposed in this chapter, the cryogenic orbital experiment was to replace the VLDS in the VDS.

Due to the fixed schedule for the proof flight which included planned ground vibrational surveys to be conducted at GDCA beginning in January 1973, a new hardware test item was identified for our program. This module is referred to here as the GVS model and was to be designed and fabricated for delivery to GDAC during the first week of tests. Design requirement for the GVS model were that its mass and dynamic properties had to be similar to the cryogenic orbital module as well as to the VLDS. In particular, no primary structural natural frequencies were to be below 40 Hz, including the effects of fluid damping on the structure.

The purpose of the GVS test was to provide data for correlation with predicted vibrational characteristics of the VLCA structure. These data would also be used for comparison with data from the proof flight. The survey was to be conducted with the VDS incorporating first the VLDS and then repeated with the GVS in place of the VLDS. In this way, the dynamic results could be

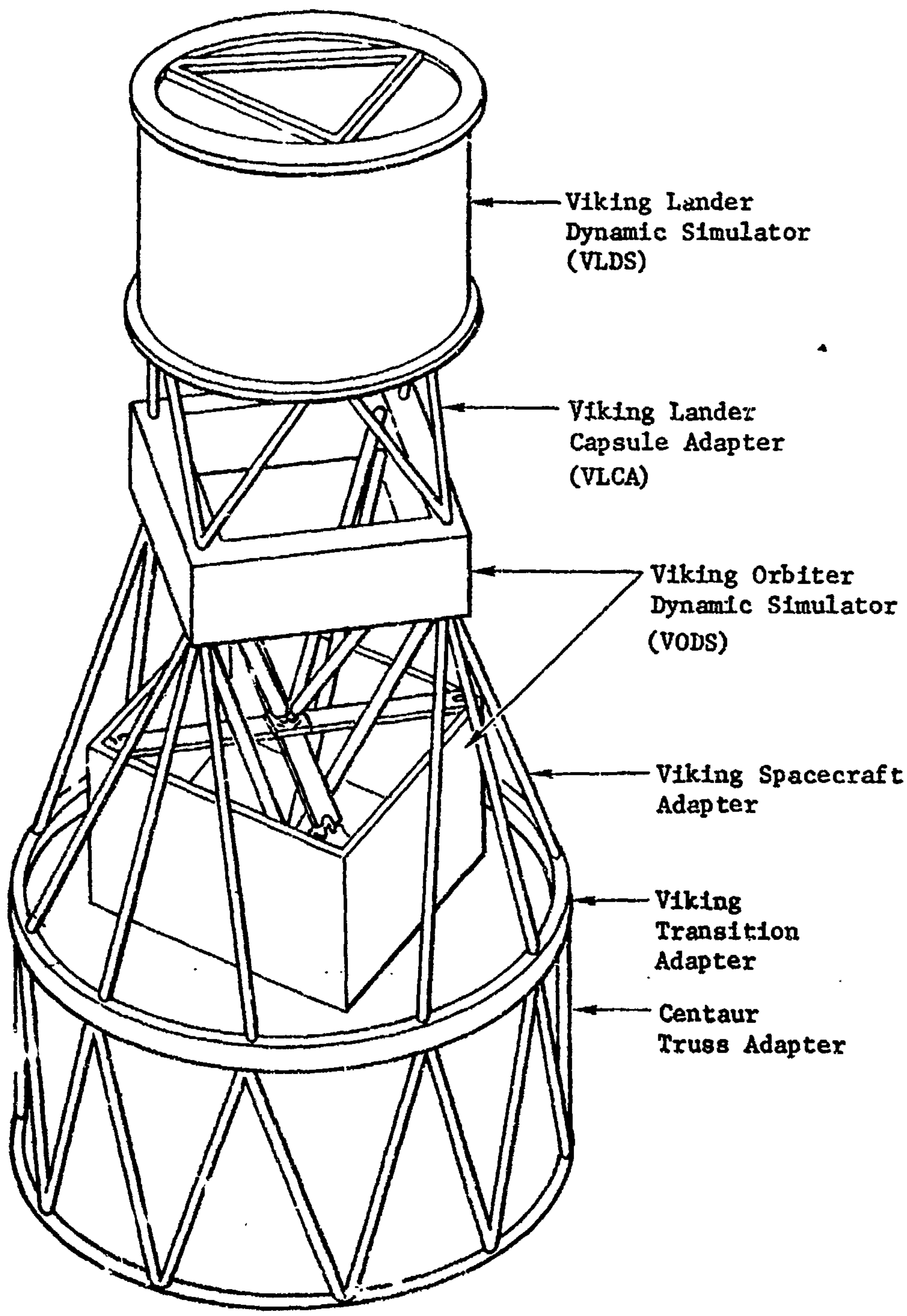


Fig. IV-1 Viking Dynamic Simulator (VDS)

compared to assure that the orbital module, assuming it was similar to the GVS, would not impair the controlled VDS flight test, as planned by NASA-LRC. Dynamic similarity between the GVS and the orbital test module was to be proven by analysis.

The VLDS is a hollow, metal drum containing no fluid; the orbital test module is a more complex structure containing LO_2. Solid body rotation, slosh and damping due to the fluid, was of major concern to LRC personnel, Ref IV-3. Differences between the GVS and VLDS due primarily to the fluid contribution, if any, and dynamic differences between the GVS and cryogenic test module were to be resolved. As mentioned, the GDCA test data would serve as a check on the analytical predictions for the VLDS and GVS difference; however, the GVS and test module were to be compared analytically only.

Aside from the dynamic similarity requirement, the principal design constraint on the orbital test module was that the primary mission objective (proof flight of the Centaur vehicle) would not be compromised. This assurance was easily provided by: (1) using large safety factors of 3.0 on yield strength and 4.0 on ultimate strength; and by (2) verifying that the lowest modal frequency of structure vibration would be 40 Hz or greater. In addition, the total mass of the package was to be 1,155 kg ± 45 kg (2550 ± 100 lb_m).

The physical size limitations were also sufficiently generous having a negligible influence on the final configuration. Lateral dimensions were restricted by the diameter of the launch vehicle shroud 4.3 M, (168 in.), and a required rattle space between the shroud and the flight test module of 22.7 cm (9 in.). This results in an envelope diameter of 3.8 M (150 in.). In the longitudinal direction there was, for all practical purposes, no restriction. As originally configured, the VLDS interfaced with the Viking Lander Capsule Adapter (VLCA) at Viking station 200.00 and extended along the longitudinal axis to station 236.00 where it interfaced with a secondary experiment, the Space Plasma High Voltage Experiment (SPHINX). Provisions for interfacing with and supporting this 79.5 kg (175 lb_m package was a requirement for the GVS model.

A. ANALYSIS

The analysis effort during the GVS article design phase was primarily in the stress and dynamic areas. The requirement that all mounted equipment and structural elements were to have natural frequencies above 40 Hz was the controlling criteria for detail design. Since overall physical dimensions and mass properties of the module were fixed, the basic structural configuration was limited. A dynamic model was generated to initially study the basic structural members. Interfaces at the VLCA and at the SPHINX were defined and a structural truss configuration was established with all experiment components treated as point masses.

The basic orbital flight test article was designed earlier in the program with consideration given to a variety of possible launch vehicles. Size was based upon providing a minimum weight experiment but large enough to produce meaningful experimental results. For the Titan/Centaur proof flight, the test article was incorporated into a heavy structural system to provide the required overall weight of 1,155 kg (2,550 lb_m). As a result, structural members were designed to meet dynamic requirements and stress levels were low. After the basic structural truss system and component locations were established, the analysis effort was concentrated in specific areas of concern.

1. Structural Truss

The truss configuration used in the analysis (Ref IV-4) is shown in Figure IV-2. The truss was 157 cm (62 in.) tall with three counterweights located at a c.g. point of 127 cm (50 in.) radius. All structural members were 7.6x7.6x0.6 cm (3x3x¼ in.) aluminum square tube and were assumed to act as axial members, i.e., no member acts as a bending element. Thus, the finite element model of the truss was the simplest to devise, but Martin Marietta experience has shown that the members in such a truss act primarily as axial members in the low modes. Local motion of the members involve bending but are modes associated with high frequencies.

The truss was assumed to be completely restrained at the lower interface since this was the governing restraint for the 40 Hz requirement. The structure of the SPHINX and the experiment components was assumed to be rigid relative to the basic porting truss (VLCA) for the GVS configuration.

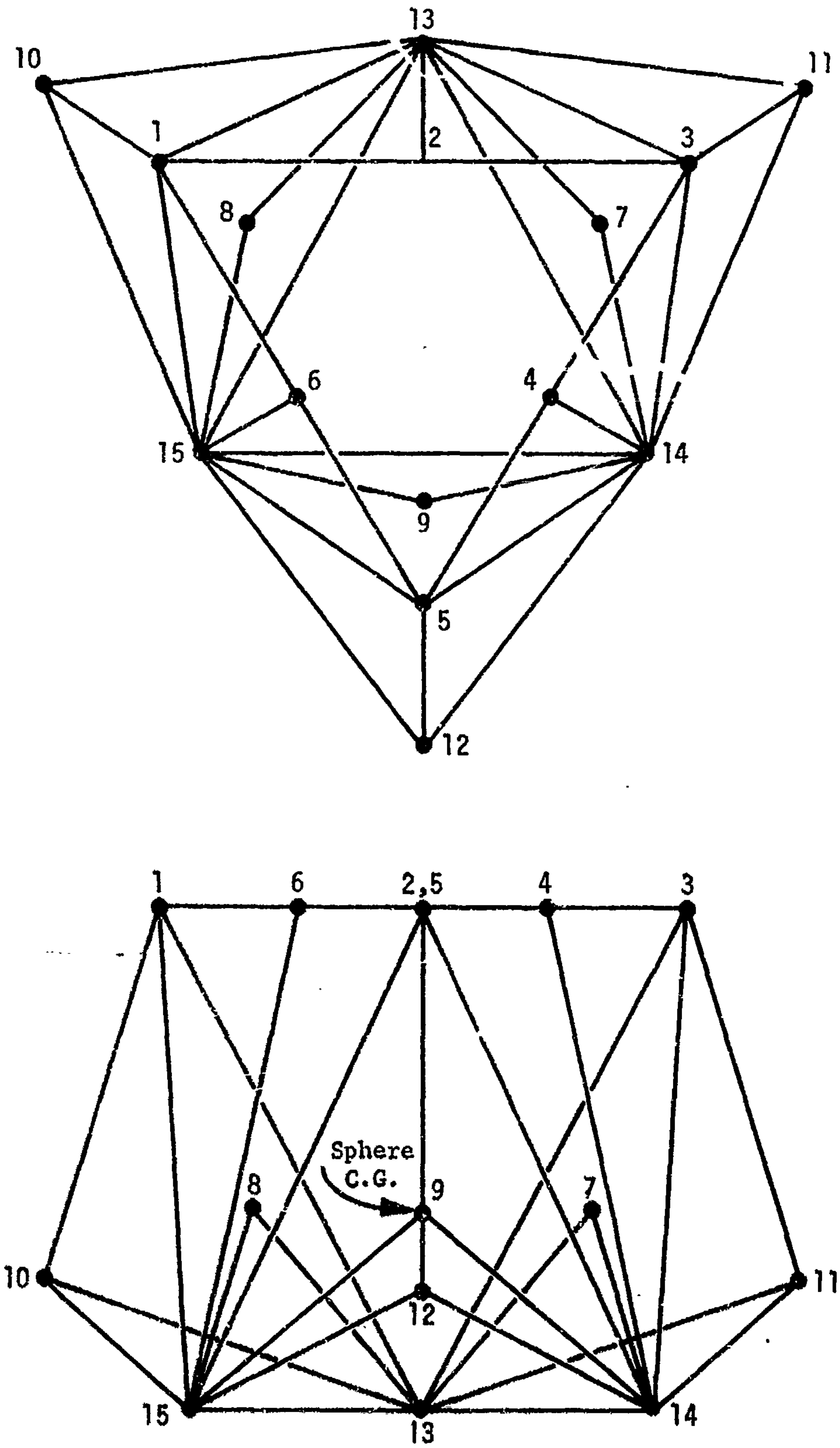


Fig. IV-2 Truss Configuration

The first four frequencies obtained in the analysis are noted in the following table with a brief description of the associated modes.

Table IV-1 Truss Modal Frequencies

Mode	Frequency	Motion
1	46.2 Hz	Outrigger/SPHINX Rocking
2	48.2 Hz	Outrigger/SPHINX Rocking
3	84.6 Hz	Axial
4	93.4 Hz	Local mode at pt. 10

The conclusion was reached that the truss configuration satisfied the 40 Hz design requirement. No difficulty was anticipated with local bending modes in the structure.

2. Gusset Plate Fatigue Analysis

Further analysis of the basic truss system with more sophisticated dynamic models (Ref IV-5) indicated a need for gusset plates to stiffen the outrigger weight arms. A typical gusset plate would cover the triangle made up of points 1, 10, and 15 in Figure IV-2. A total of six gusset plates were needed to cover all affected areas.

The addition of these gusset plates presented a potential acoustic vibration problem. Large expanses of material are susceptible to failure due to acoustic loading. Such failure may occur by: (1) acoustic pressure loading creating stresses exceeding the strength of the material; or (2) acoustic pressure loading causing fatigue after a given time even at low induced stress levels. For the analysis the gussets were assumed to be 0.63 cm (0.25 in.) aluminum sheet in triangular configurations with two edge support conditions. The acoustic loading used was the anticipated Titan IIIE/Centaur launch lift-off level of 145 dB.

The analysis results indicated the acoustic loading was in no case sufficient to produce stress beyond the material strength. The possibility of fatigue failure was analyzed by calculating

the natural frequency of the gusset configurations and edge support conditions and by using the appropriate S/N data for the material. A computer program was used to perform the numerical integration required to predict the time to failure. In all cases, the time to failure was found to be much greater than the time span of the acoustic input. At this point the basic structural system analysis was complete, with a high degree of confidence, in meeting the dynamic and stress requirements.

3. Tank Liquid Slosh

The orbital experiment liquid oxygen tank consisted of a vacuum jacketed, multilayer insulated spherical tank containing a dual-screen-liner (DSL) propellant acquisition device. Support of the tank within the vacuum jacket was provided by a system of rods to minimize heat loss to the liquid oxygen. The GVS simulated tank, however, was a single spherical tank of the same diameter, 91 cm (36 in.), as the vacuum jacket. It was to be filled to approximately 5% ullage with kerosene or fuel oil to provide the same total mass as the experiment tank. Since the GVS was to be as dynamically similar to the flight article as possible, considerable concern was expressed over the propellant tank similarities.

The first analysis conducted (Ref IV-6) involved the masses and frequencies of liquid slosh. The following table lists the result of the analysis for the GVS tank.

Table IV-2 Slosh Frequencies

% Ullage	Frequency - Cps	% of Mass in Slosh
5	1.45	9.2
10	1.29	13.5
15	1.22	15.6
50	0.97	17.5

As indicated in Table IV-2, at the 5% ullage loading, a mass of 9.2% of the propellant total mass will be creating slosh forces at a frequency of 1.45 Hz.

4. Tank Support Analysis

A dynamic model was generated to analyze the experimental tank

internal support system. Several cable and rod support configurations between the tank and the vacuum jacket were analyzed (Ref IV-7) and a system of nine support rods was selected. This system was made up of three groups of three rods, at three points on the inside of the vacuum jacket, at the horizontal girth ring. The rods radiated from the vacuum jacket to the propellant tank through three equal angles. The first natural frequency, considering the support system fixed at the vacuum sphere inner surface and the inner tank rigid, was 48.6 Hz. The rods used for this analysis were assumed to be 0.32 cm (1/8 in.) dia by 19 cm (7.5 in.) in length and were 304 stainless steel.

When the total system was considered (i.e., the internal supports and the bipod support of the tank on the module structure) a coupling problem was evident. The bipod system frequency was calculated to be 126 Hz. To maintain a minimum frequency of 40 Hz for the overall system, an internal tank support frequency of 70 Hz was required, resulting in an increase in the support rod diameter from 0.32 to 0.46 cm (1/8 to 3/16 in.) for stainless steel or a titanium rod diameter of 1.43 cm (9/32 in.). The dynamic analysis also provided 1 g structural loads in the rod supports.

With no tension preload, maximum loads of 548 kg (1205 lb_m) in compression were obtained. The stress levels under such load would be 30 x 10^7 N/m^2 (43.6 ksi) for the stainless steel rods. The tension preload in the rod members should exceed the maximum load limit to assure that the rods always act in tension and the possibility of a buckling failure is eliminated.

5. Structural Damping

Discussions were held with NASA-LRC to review the orbital experiment/GVS dynamic comparison. The LRC dynamic group was the sponsor of the flight dynamic study of the VLCA and consequently was the key to final acceptance of the experiment for flight. An item-by-item review of the orbital experiment/GVS systems indicated no concern with the structure or equipment components; however, considerable concern was expressed with respect to fluid damping similarities. Fluid damping and the resultant force feedback into the structure can be considerably different for the two tank configurations. The problem was to determine the magnitude of this difference and its significance with respect to total structural damping.

This problem involved the difference in forces fed back into the tank structure due to slosh damping on the screen liner in the experiment tank, and slosh damping on the tank wall in the GVS tank. The screen device will be completely wetted at the small ullages considered (5% or less) and may trap an ullage bubble within. In effect, the wetted screen will then become a barrier to fluid flow and sloshing will occur only in the annulus between the screen liner and the tank wall.

An analysis was begun to determine the fluid damping forces in the experiment tank. The approach taken was to compute flow coefficients for liquid flows between the tank wall and the screen liner. This included estimates for flow perpendicular to the flow channels. At the same time, an effort was made to obtain the mode shapes and frequencies from the structural damping model runs at JPL. The intent was then to apply the fluid damping feedback, based on flow coefficients calculated, to the overall structural damping model. Depending on the results of this analysis, a plan would be formulated to follow one of the following courses of action:

1. Proceed without change because the feedback forces due to liquid damping are not significant.
2. Design a partial screen liner or baffle system for the GVS tank to provide the required similarity.
3. Subject the orbital experiment to a ground vibration survey at KSC after fabrication completion. This test would occur in the September - November 1973 time span.

At the time of program termination this analytical effort was just beginning and no results had been obtained.

B. DESIGN

The shape of the GVS structure was defined, to some extent, by the interface configuration of the VLC adapter at the bottom of the structure and by the SPHINX interface configuration (Ref IV-8) at the top. Each interface configuration consisted of three equally-spaced, support points: (1) the VLC adapter points (station 200.0) located on a 152 cm (60 in.) dia circle and the SPHINX support points located on a 87 cm (34.25 in.) dia circle. The distance between these two triangles was determined by the space requirements of the experiment's liquid oxygen tank assembly and by the SPHINX. Although the SPHINX is supported at

station 263.13, its cannister assembly extended below that station approximately 40.6 cm (16 in.). The oxygen tank assembly of the new secondary payload was to be simulated in the GVS model with a 91.4 cm (36 in.) dia aluminum sphere as shown in Figure III-7.

The large safety factors and the relatively high vibrational frequency requirements dictated the use of heavy structural members. Conventional airborne design would normally specify aluminum tubing with a 5.08 cm (2 in.) square cross-section and wall thickness of 0.124 cm (0.049 in.) for the structure. By contrast, the GVS design specified the use of aluminum tubing with a 7.62 cm (3 in.) square cross-section and a wall thickness of 0.63 cm (0.250 in.).

While the interface mounting plane at the top of the VLC adapter was at Viking station 200.00, the centerlines of the adapter struts intersected at three points in the plane at station 204.28, as shown by Figure III-7. The apexes of the GVS base triangle were coincident with these points. Three mounting pads were to be welded to the base triangle to act as spacers and to provide the required interface hole pattern for vehicle assembly.

The upper triangle of the GVS structure was rotated 60° with respect to the base triangle so that the two triangles formed parallel planes with their centers on the longitudinal axis. Unless otherwise specified, points on the structure were designed as the intersection of beam centerlines. The upper triangle was supported by nine struts which emanated from the base triangle, three from each apex. These struts were designed to terminate at the upper triangle so that there were two at each apex and one at each of the three points midway between. These three midpoints were also the focus of the SPHINX support pads; thus providing a direct load path to the VLS support points.

At this point in the design, it was necessary to identify the experiment components and to decide which had sufficient mass to require simulation on the GVS. These components are identified below.

Simulated on GVS	Not Simulated
LO_2 tank assembly	Antennas
GOX pressurant tank	Thermal blankets
Batteries (2)	Cabling
Instrumentation pod	Pressurant plumbing
Command link pod	Vent plumbing
Feedline	Transducer capsules

The LO_2 tank assembly that would be used on the actual flight model test module consisted of: (1) the pressure vessel; (2) vacuum jacket; (3) capillary liner; (4) insulation; (5) instrumentation; and (6) liquid oxygen. The total weight of this assembly was estimated to be 341 kg (750 lb_m) with 252 kg (556 lb_m) contributed by the liquid when loaded to 5% ullage. The aluminum tank designed for the GVS included a basic sphere weight of an additional 5.4 kg (12 lb_m) to account for the girth ring, for a total of 34 kg (76 lb_m). To bring the tank assembly weight up to the required 340 kg (750 lb_m), the contents of the tank were required to weigh 305 kg (674 lb_m); this was accomplished by filling the tank to 5% ullage with a liquid having a specific gravity of approximately 0.8. Diesel fuel or kerosene were acceptable liquid candidates. However, these liquids did require a leak-tight tank to avoid a safety hazard during the ground vibration testing.

The tank was designed to be supported at three equally spaced points on its girth ring by three bipods which were themselves supported by the base points. The initial tank configuration made no provision for any internal structure simulating the dynamic properties of the capillary liner, pressure vessel, or suspension systems. This omission was identified as a possible problem and was subjected to analytical evaluation. Fabrication of the GVS was stopped before the results of this evaluation became available.

The required moment of inertia about the longitudinal axis dictated that some, if not all, of the remaining components be positioned well away from that axis. Because the basic structure had already taken on a three-sided configuration, the three heaviest components were selected for tripod-type mounting outboard of the central structure. The design provided for mounting the GO_2 pressurant sphere and the two batteries in this manner.

Since gaseous oxygen had been selected as the pressurant medium, a titanium vessel could not be used. The quantity and storage pressure of the pressurant required a stainless steel sphere 38.1 cm (15 in.) dia with a wall thickness of 1.91 cm (0.75 in.). For the GVS model this tank was simulated with a stainless steel sphere of the same diameter but with a wall thickness of 0.16 cm (0.063 in.). The disparity in weight was made up by the use of 2.5 cm (1.0 in.) thick plate through the girth plane and elsewhere

inside the sphere. The sphere was mounted on a stainless-steel plate using four equally spaced stainless-steel-angle bipods. The design weight of this welded assembly was 98 kg (216 lb).

Although each of the batteries weighed substantially less than the GO_2 tank assembly 70 vs 98 kg (154 vs 216 lb_m), all three of these components were positioned equidistant from the longitudinal axis in the interest of maintaining structure simplicity. This distance was approximately 1.4 M (55 in.) measured to the center of the sphere or to the c.g. of the battery. The point of intersection of each of the supporting tripod members was, however, at a distance of 1.26 M (50 in.) from the longitudinal axis to coincide more closely with the c.g. of the sphere/mounting plate assembly and the battery/battery-box assembly. In this connection, each battery box contained a volume into which a lead ballast could be placed as required in the final balance.

The instrumentation pod 47 kg (104 lb_m) was designed to be placed inside the structure adjacent to one of the batteries while the command link pod 16 kg (35 lb_m) was to be placed near the other battery.

To allow draining of the liquid from the 91 cm (36 in.) sphere, a 1.9 cm (3/4 in.) aluminum pipe was attached to an elbow and valve at the bottom of the tank and routed to a point beneath the GOX sphere at station 201.30. This assembly did not exactly simulate the weight of the flight article feedline, but occupied the position of the orbital test module. The GVS model design was completed with the installation of a fill valve and pressure relief valve at the top of the tank. These valves were for convenience only and had no counterparts on the flight article.

C. FABRICATION

Fabrication of the GVS article was initiated and progressed to approximately 80% completion when the program was terminated. The partially assembled GVS is shown in Figure IV 3. Major component simulators such as the batteries, pressurant tank, electronic boxes, and the basic truss structure had been completed.

The liquid oxygen tank and its supporting structure had not been completed because of the questions raised concerning the degree of dynamic simulation of the orbital test module tank.

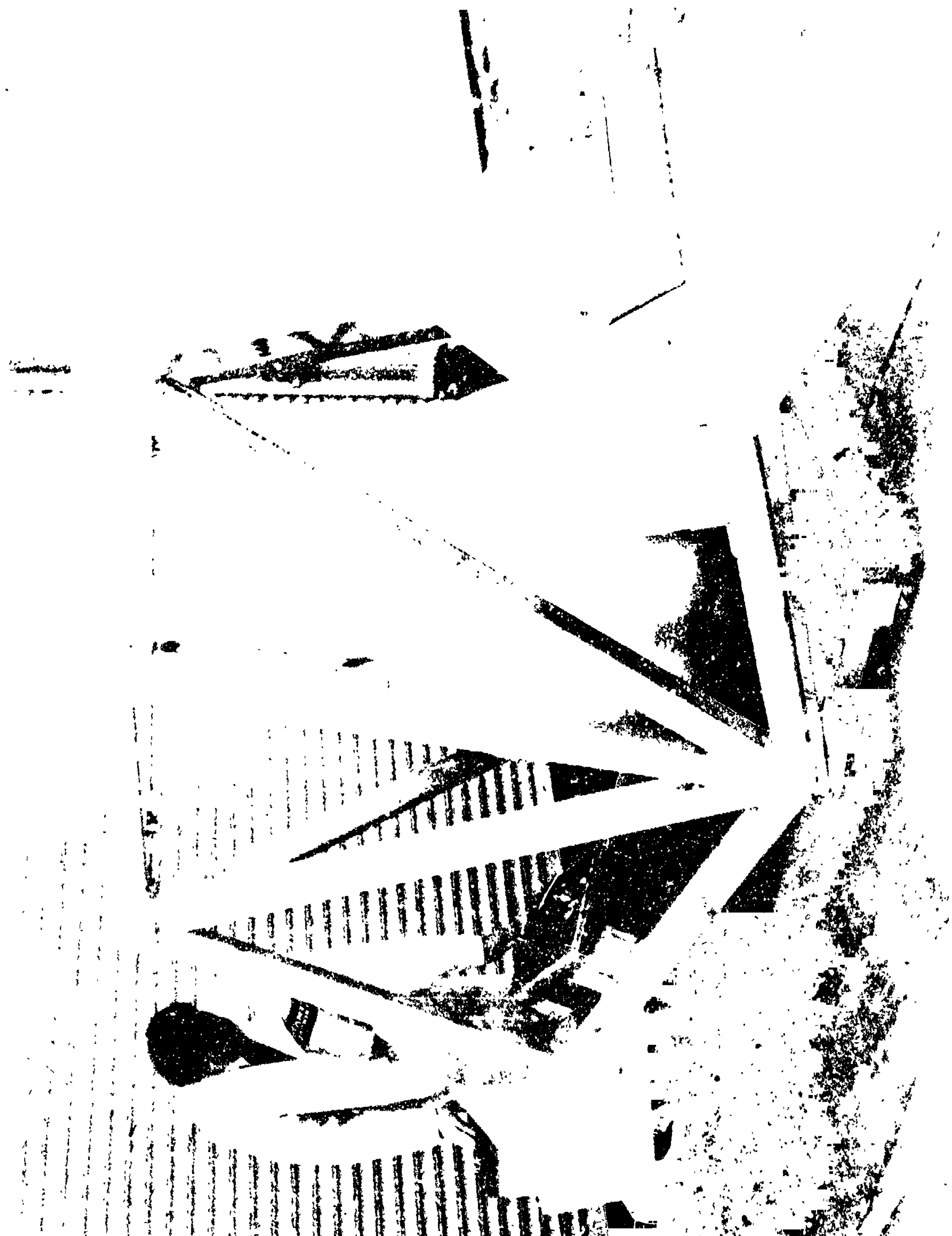


Fig. IV-3 Partially Assembled GVS

The basic truss system was made of welded aluminum box sections with a removable section provided in the upper triangle to allow installation and removal of the LO_2 tank. The battery simulators and the electronic box simulators were built of aluminum plate stock welded or bolted together. The pressurant sphere was stainless steel and was ballasted internally with stainless steel plate to provide proper mass simulation. Space was provided in the battery support boxes for lead ballast to allow final trim of total mass and moments of inertia after GVS fabrication.

TEST

After completion of the GVS article, several tests were planned prior to flight of the orbital test module. The first series of tests were preliminary resonant survey tests to be conducted in Denver. These tests involved mounting the GVS on a rigid base support such as a concrete pad. Accelerometers were mounted on components or structural members where lowest resonant frequencies were expected. The assembly would then be stimulated or caused to vibrate and accelerometer outputs recorded to verify primary mode natural frequencies. This elementary test would help to verify the resonant frequency analysis and would allow test article rework if required prior to shipment to GDCA for the payload stack modal survey tests.

The second series of tests, which would have occured at GDCA, would primarily involve testing of the VLCA. The test would be conducted with the complete proof flight payload. Accelerometers would be mounted on the various payload segments, including the orbital test module. The entire payload stack would be vibrationally stimulated at the base. Accelerometer readouts would be observed for resonant frequencies. The objective of this test was to observe the dynamic characteristics of the VLCA and to compare the test data with data obtained during the proof flight. With VLCA natural frequencies in the range of 5 to 15 Hz, the remainder of the payload was required to have natural frequencies above 40 Hz to avoid coupling. This test program could result in changes to the VLCA or to the simulated payload and resulting retest with a long schedule span possible.

At the completion of the testing at GDCA the GVS program would be complete. The VLDS would be shipped to KSC for fit check and road testing of the proof flight payload. The simulator would

then be stored at KSC until the launch of the proof flight vehicle. In this way the VLDS would be available as a backup for the orbital test module in case a problem occurred that precluded flight of the experiment. This procedure was consistent with the requirement that the orbital test module must not compromise the primary objectives of the proof flight.

V. CONCLUSIONS AND RECOMMENDATIONS

The passive DSL tank/feedline design is an extremely attractive system for many propulsion, life support, auxiliary power, and cooling applications where efficient subcritical cryogen storage during low-g is desired. As described in Volumes II and III, a significant analytical and test effort has verified the DSL performance while characterizing certain operational constraints. However, as concluded in Volume III, an extended low-g test is needed to qualify this advanced concept for incorporation into future Earth-orbiting vehicles and payloads.

During the past ten years, numerous studies on the use of various capillary devices for non-cryogen fluid control have been performed. A partial list of these studies appears in Refs V-1 through V-12. Bench testing under 1 g and limited low-g periods, provided in drop tower tests and KC-135 airplane flights, have provided considerable qualitative data applicable to the DSL.

Under Contract NAS9-10480, Martin Marietta delivered a DSL model that was flight tested in November, 1971, in the KC-135. This tie-down test, using methanol as the test liquid, demonstrated gas-free liquid expulsion during the 25-sec low-g period. This time period is too short to fully evaluate the thermal effects of long-term storage of cryogens. As an example, the time required to establish a steady, free convection boundary layer flow during low-g can be on the order of 18 hr or more. If a convective boundary layer does not form, appreciable radial temperature gradients tend to result from purely conduction dominated heat transfer. The DSL provides a vapor annulus between the bulk liquid and storage tank wall to minimize these undesirable temperature gradients. Testing of the 63-cm (25-in.) dia LH_2 spherical tank (described in Volume III) experienced this temperature stratification that was worsened by the use of LH_2 and the 1 g environment. Vapor-free LH_2 outflow was successfully demonstrated, as was the communication (gas annulus-to-bulk region) performance for the total liner. However, liquid-free vapor venting could not be demonstrated because of the 1 g stratification phenomena.

In summary, it is recommended that the next step required to verify the DSL concept is a long-term (on the order of days) low-g test to demonstrate satisfactory vapor venting, liquid cryogen outflow, and bulk fluid control. Two program plans for placing a cryogenic flight test article into an Earth orbital flight are described in this

volume. The secondary payload approach (Section III) is recommended because of its lower cost. Either the secondary or the dedicated payload scheme is acceptable, however, to validate the DSL design in the desired 7 to 14 day low-g period. Such an orbital test effort is required circa 1974-1975 if this promising design is to be incorporated into future vehicles, such as Space Tug and other Shuttle payloads that will be operational in 1979.

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